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STRESS CORROSION OF TITANIUM ALLOYS UNDER SIMULATED SUPERSONIC FLIGHT CONDITIONS

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LOCKHEED-CALIFORNIA COMPANY
Burbank, Calif.

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SUMMARY

The purpose of this program was to determine if the titanium hot salt cracking process is significantly affected by the presence of high altitude supersonic air flow conditions. Testing was performed in Lockheed's Supersonic Wind Tunnel Facility. Altogether, a total of 50 hours of exposure time under Mach 2.5 supersonic flow conditions at a pressure equivalent to 70 000 ft altitude and temperatures of approximately 600°F and 700°F was accomplished. Two different types of self-stressed specimens were utilized in these tests; one type was developed by Lockheed and the other by NASA Langley. Specimens were prepared from both the Ti-8Al-1Mo-1V duplex annealed and the Ti-6Al-4V mill annealed alloys. Cracking data from specimens exposed in the wind tunnel were compared to similar data obtained from ordinary laboratory oven tests. The tests demonstrated that the air flow and reduced pressure conditions characteristic of high altitude supersonic flight environments do not eliminate hot salt cracking in these titanium alloys at the test temperatures used in this program. However, cracking in supersonic environments was generally less severe than in laboratory oven environments. As might be expected, some of the salt coating was removed from the specimens by the action of the high velocity air stream, and evidence of a correlation between salt removal and cracking damage was indicated.

Section 1

INTRODUCTION

In the past, titanium stress corrosion cracking has been studied in many laboratory testing programs and there is a great deal of information available concerning this phenomenon. Experiments have been conducted under varying conditions of temperature, exposure time, atmospheric environment and alloy composition. In addition a few tests have been reported in which the effects of atmospheric motion on the corrosion process were investigated. However, with a single exception of all these past programs were performed under ordinary laboratory conditions using relatively low air velocities. The only previous program which has attempted to simulate supersonic flow conditions was reported by Heimerl (Ref. 1). In that study, self-stressed salt coated specimens were subjected to a Mach 3 air stream at 460°F for several 25 second runs. This was far too short an exposure to produce any measurable corrosion or cracking. Much of the salt was removed by the high velocity air, but enough remained to produce cracking upon subsequent exposure in a laboratory oven.

The obvious lack of hot salt corrosion data under simulated flight conditions prompted researchers at Lockheed to propose this program. The basic objective of this research effort was to perform titanium hot salt corrosion tests in a simulated high altitude supersonic flight environment, and to compare the results of these tests with those obtained from specimens exposed under similar conditions of time and temperature in ordinary atmospheric pressure laboratory ovens.

In this investigation two different alloys (Ti-6Al-4V mill annealed and Ti-8Al-1Mo-1V duplex annealed) were used. Simultaneous testing was performed using two distinctly different types of self-stressed specimens. One type is the well-known constant bending moment specimen developed by NASA Langley; the NASA configuration produces a uniform tensile stress over the entire curved outer surface of the specimen. The other type of specimen tested was developed at the Lockheed-California Company. The Lockheed bend specimen consists of a small strip of metal in a buckled configuration. This type specimen provides a whole range of known tensile stress levels in the outer surface of the material. Typical specimens of both types are shown in Figure 1-1.

The NASA and Lockheed specimens yield somewhat different types of information. After exposure, the NASA specimens are subjected to a special compressive loading test which provides important information concerning the loss of bending ductility and mechanical strength which has occurred during the stress corrosion process. On the other hand, cracking damage on the Lockheed specimens is assessed by direct microscopic observation of the metal surface, providing information concerning the numbers and lengths of the cracks which have occurred. In addition, the Lockheed method allows the determination of the minimum stress level required to produce cracking in a given specimen (the stress threshold value).

The Lockheed specimens were prepolished to provide a sufficiently smooth surface to permit cracks to be observed under the microscope. The NASA specimens were tested in the as-received surface condition. All specimens tested (both Lockheed and NASA) were prepared from the same mill sheet stock starting material.

Three separate exposure tests were performed in the wind tunnel. The first two runs of 15 hours each were made under conditions of 700°F specimen temperature, Mach 2.5 air flow and 70 000 ft pressure altitude. An additional 20 hour test was performed under similar conditions of pressure and airflow, but the specimen temperature was lowered to approximately 600°F. A testing schedule was established so that certain specimens were removed at the end of each run and replaced with new ones. Thus, specimens were obtained with total accumulated exposure times of 15, 20, 30, and 50 hours.

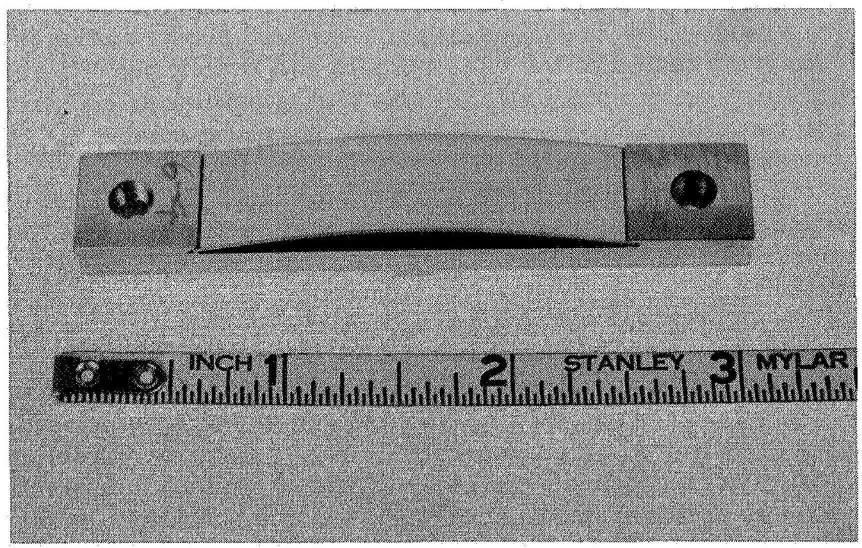


Figure 1-1a LOCKHEED BEND SPECIMEN IN HOLDER

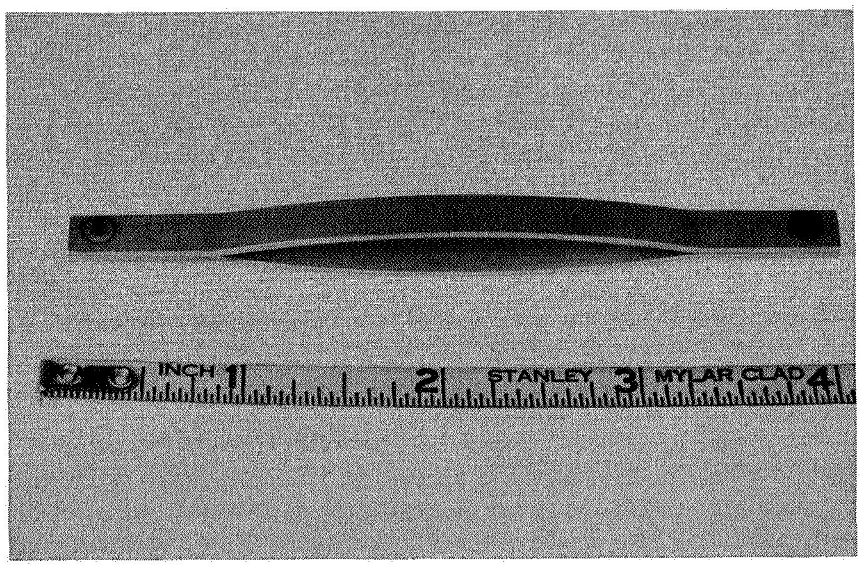


Figure 1-1b NASA SELF-STRESSED SPECIMEN

Altogether, a total of 384 specimens was tested in this program, half of which were prepared by NASA Langley in the NASA configuration, and the remainder were prepared by Lockheed in the Lockheed configuration. Of this this number, 112 were tested in the wind tunnel; the remaining 272 were exposed in laboratory oven tests. Of the oven test specimens, 176 were coated with salt and the remaining 96 were tested in the uncoated condition. The purpose of testing the uncoated specimens was to provide base line data. Two distinctly different types of laboratory oven tests were performed. In one type of oven test, the time-temperature history of the tunnel test specimens was duplicated as closely as possible. In the other type of oven test, specimens were exposed under constant temperature conditions representative of the average equilibrium temperature of the tunnel test specimens.

To ensure that a measurable amount of stress corrosion cracking would occur during the relatively short tunnel exposures, the test specimens were designed to produce relatively high outer fiber stress levels. The combination of the high exposure temperatures and stress levels resulted in a significant amount of stress relaxation in the specimens. Estimated stress relief values as high as 10 KSI in the 600°F specimens and 26 KSI in the 700°F specimens were observed. Allowances for this stress relief effect were made in the interpretation of the test results.

There were some variations in the outer fiber tensile stress levels of the specimens used in the various tests. These variations were the result of such factors as the basic difference in the elastic moduli of the 8-1-1 and 6-4 alloys, the dependence of the moduli on temperature, and slight variations in specimen geometry. Due to these factors the average initial maximum stress levels in the Lockheed 6-4 alloy specimens were approximately 75 KSI at 700°F and 84 KSI at 600°F. For the NASA 6-4 specimens the initial stress levels were estimated to be about 76 KSI at 700°F and 80 KSI at 600°F. For the Lockheed 8-1-1 specimens the initial maximum stress levels averaged about 78 KSI at 700°F and 83 KSI at 600°F, whereas for the NASA 8-1-1 specimens the stresses were approximately 86 KSI at 700°F and 90 KSI at 600°F.

After testing, all NASA specimens were returned to NASA Langley for evaluation whereas the Lockheed specimens were processed at Lockheed. The results of these investigations are presented in this report. Pertinent time-temperature history data and post-test photographs of all NASA and Lockheed tunnel test specimens are also presented.

It is believed that the information gained in this program will be of value in determining whether stress corrosion cracking data obtained in ordinary laboratory oven tests are useful for predicting cracking behavior in high altitude supersonic environments.

Section 2

HOT SALT CORROSION TEST TECHNIQUE

Lockheed Specimen Configuration

Various methods of studying hot salt corrosion have been investigated at Lockheed in recent years, and a technique was developed which yields comparative stress corrosion cracking data in an efficient manner without requiring massive, dead weight loading. Furthermore, the small specimen size used in this test method ensures that a large number of specimens can be tested simultaneously in a limited space. Hence, it is ideal for use in tunnel testing where space is at a premium and the cost of utilizing auxiliary tensile loading equipment would be prohibitive.

The basic·bend specimen used in this method is a sheet of optically polished titanium which is buckled and placed in a special stainless steel holder (Fig. 1-1a). For this specimen, tensile stresses are produced in the outer fibers and compressive stresses in the inner fibers. This specimen has an important advantage in that it provides a whole range of known stress levels from essentially zero at the ends of the specimen to a maximum value at its center. The stress distribution as a function of distance along a typical specimen is shown in Fig. 2-1. The effect of stress level on the cracking process may be judged by comparing the extent of cracking in different regions of the specimen surface, and relating these data to the particular stress levels involved. An additional benefit gained by the use of this method is that the minimum stress level required to produce cracking (the stress threshold) may be estimated by determining the locations of the last cracks toward the ends of specimen and relating their positions to the stress level in that region.

There is a significant advantage in using optically polished titanium specimens in these tests. In general, it is difficult to directly observe minute cracks on the surface of commercial titanium because of the masking effect of normal surface irregularities. When prepolished specimens are used, however, cracking and corrosion may be clearly observed under the microscope at 400X magnification. In addition, a special etch was used on these polished specimens to further reveal cracking damage. This etch (2 cc HF, 12 cc H_2O_2, and 6 cc H_2O for 10 seconds) was developed by NASA Langley for detecting cracks in unpolished specimens. When this etch is applied to polished specimens, the shiny background metal remains essentially unaffected whereas the cracks become very dark and easily observable. This etch proved to be of great benefit to the microscope observers who obtained the cracking data. Incidentally, microhardness measurements have demonstrated that the surface polishing procedure does not produce measurable work hardening of the metal surface.

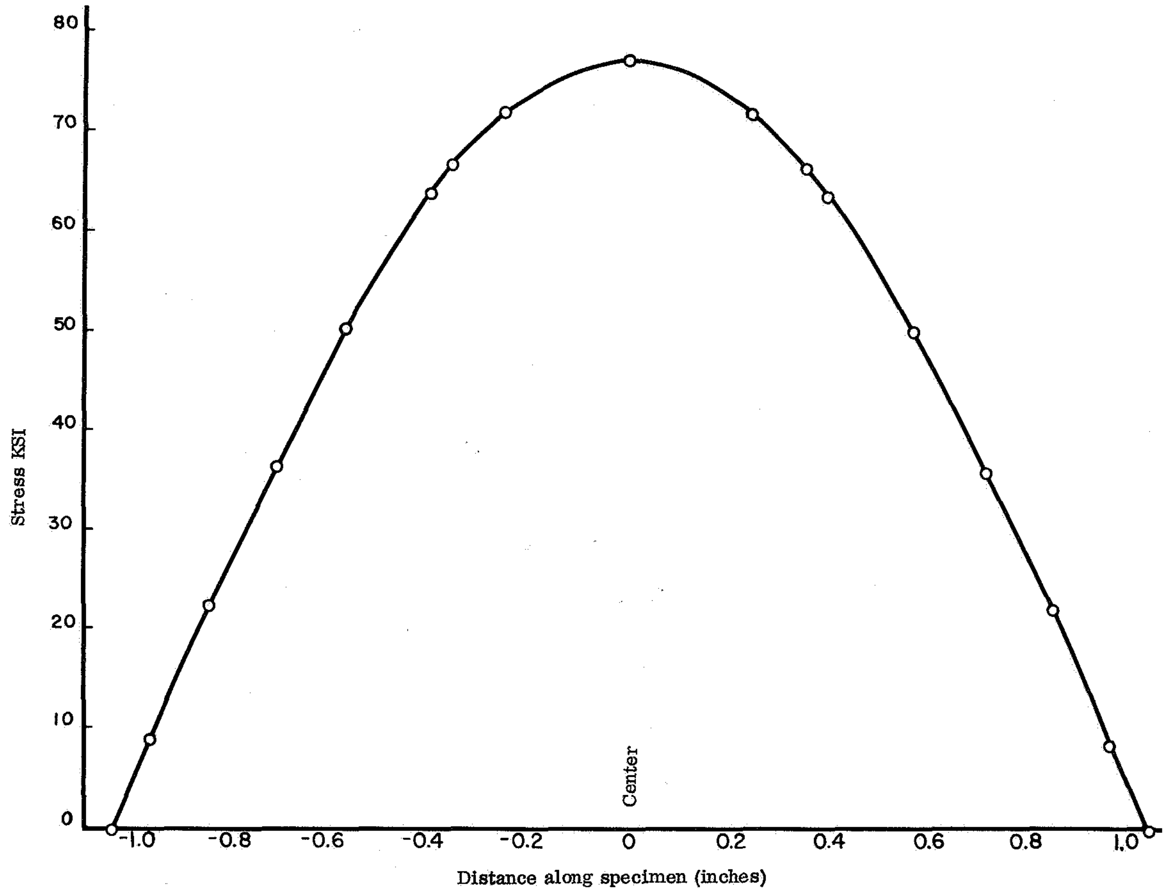


Figure 2-1 OUTER FIBER TENSILE STRESS DISTRIBUTION FOR A TYPICAL LOCKHEED BEND SPECIMEN

Outer fiber stress level may be computed at any point along the specimen length by means of an analytical mathematical procedure. Using this procedure a stress distribution curve (similar to Fig. 2-1) may be generated for each test specimen. Workable mathematical relationships describing the stress condition of the Lockheed specimen are provided in the Appendix of this report. Also included in the Appendix is a method of estimating the magnitude of stress relief which may have occurred in these specimens during the elevated temperature exposure.

Lockheed Specimen Preparation

The initial step in preparing the Lockheed test specimens was to cut the 0.040 inch thick titanium sheet stock into approximately 2 x 2 1/2 inch rectangular pieces. The individual pieces were mounted to an end of a brass disk 3 inches in diameter and 1 1/2 inches thick using a transparent double-sided adhesive tape. The titanium pieces were allowed to run on a 12 inch lapping machine using 5 micron alumina abrasive in lapping oil on a serrated cast iron lapping plate. The lapping process continued until the thickness of the titanium was reduced to about 0.030 inches. At this point the metal was removed from the lapping machine and prepared for the polishing operation. The entire lapping operation required about 4 hours. It was necessary to reduce the thickness of the titanium to about 0.030 inches since specimens prepared from thicker stock would have produced bending forces which were unmanagably large. Figure 2-2 shows a brass mounting disk being placed on the lapping machine.

Immediately after the lapping operation, excess lapping oil and abrasive were removed from the titanium by a several minute treatment in a ultrasonic cleaner containing methyl ethyl ketone solvent. After being thoroughly cleaned, the lapped titanium was transferred to a rotary metallurgical polisher using a soft cloth with 0.3 micron alumina in water at 264 rpm. The polishing operation was continued until no surface pits or other irregularities were visible at 400X magnification. After about an hour a mirror finish was achieved.

After the polishing operation, the sheet was cut into 1/2 inch wide strips on a water-cooled abrasive wheel. The ends of each strip were then precision ground to the required length and degreased for several hours in a vapor phase cleaner containing methyl ethyl ketone solvent. The strips were then inserted into stainless steel specimen holders. For this operation the technician wore disposable polyethylene gloves.

Salt coating was accomplished by spraying a saturated salt solution on the specimens which had been warmed slightly above room temperature to hasten evaporation. All Lockheed wind tunnel specimens were coated in place in the wind tunnel*. The tunnel specimens were warmed by directing hot air on them from an electric heater. All oven test specimens were coated on a hot plate which was maintained slightly above room temperature to promote evaporation. The coating process was repeated until a salt thickness corresponding to about 0.003 inches was attained. Coating thickness was determined by monitoring the weight increase of representative control specimens.

After being tested, the specimens were washed in deionized water to remove the salt and wiped lightly with alumina powder in water. This process removes excess corrosion products and virtually restores the surface polish. Then the

*NASA specimens were precoated with salt at NASA Langley before being shipped to Lockheed.

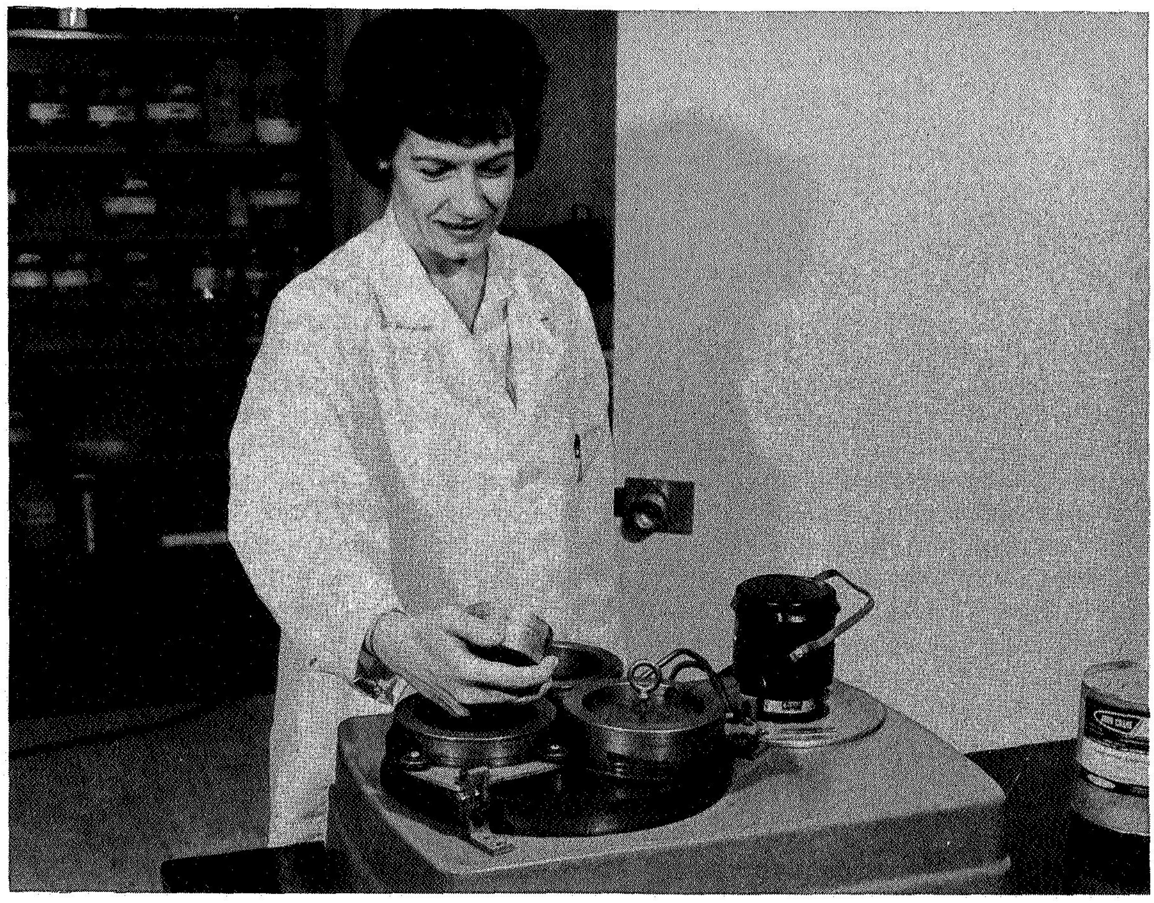


Figure 2-2 MOUNTED SPECIMEN BEING PLACED ON LAPPING MACHINE

specimens were given the etching treatment described earlier. This etching process causes the cracks to be preferentially attacked and renders them readily visible when contrasted against the polished background. It is a fairly simple task to locate the position of cracks on a precision stage microscope and quantitatively relate these to the particular stress levels involved.

Lockheed Specimen Data Analysis

The utility of the Lockheed test technique presented here can be illustrated most satisfactorily with some actual test results. A typical 6-4 alloy specimen was chosen from those tested and used as an example to demonstrate this test method.

Microscopic examination of the specimen revealed that rather extensive cracking occurred in regions of highest tensile stress (near the center of the specimen). The extent of the cracking decreased with decreasing tensile stress and all signs of cracking vanished at a rather sharply defined distance from the specimen center, indicating that there is a critical stress threshold level below which no cracking occurs. The existence of this threshold is best demonstrated by means of a crack severity study. Cracking severity is determined by viewing the prepared specimen at 400X magnification with a series of parallel grid lines superimposed on the crack network as observed in the microscope field (Fig. 2-3). Each time a crack crosses a grid line, a single count is made; the number of counts observed in a square millimeter of surface area is a measure of the severity of cracking. Notice that a single crack may give rise to several counts since it may cross several grid lines. Thus, the crack count data are dependent on both the numbers of cracks present per unit area, and the individual lengths of these cracks. For the specimen described here a series of crack intersection counts were made in narrow increments located in the center of the specimen, and at 5 mm and 10 mm on either side of center. In addition, the positions of the final cracks on either side of center were determined. Since the specimen is symmetrical about the center, corresponding data on either side were averaged. This averaging process helps to compensate for minor data variations resulting from unavoidable inconsistencies in specimen thickness, width, composition, etc.

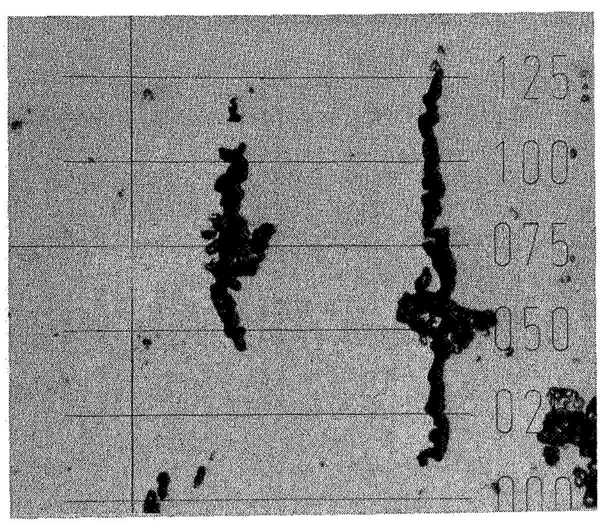


Figure 2-3 APPEARANCE OF ETCHED HOT SALT CRACKS AT 400X MAGNIFICATION (SUPERIMPOSED GRID LINES ARE APPROXIMATELY 0.001 INCHES APART)

The stress level at each increment was determined utilizing the mathematical relationships described in the Appendix. The crack count data were then plotted as a function of stress level. A plot of this type for the example 6-4 alloy specimen is shown in Fig. 2-4. Notice that the relationship of count data to stress level data may be fairly well represented by a straight line type correlation.* If desired, the position of the line may be established by applying a least squares type analysis to the data. The point at which the line intercepts the horizontal axis defines the stress threshold value for the specimen. This critical stress level represents the maximum tensile stress which may be applied to a specimen without producing cracking under the particular exposure conditions.

Notice in Fig. 2-4 that the slope of the data line is indicated. The line slope is a direct measure of the extent of the cracking damage which occurred in a particular specimen and is a direct indication of the effect of stress on the cracking process. The actual physical significance of the line slope may be better understood by examining the dimensions of this quantity which are counts/MM^2/KSI. Thus, a line slope of 3.7 as shown in Fig. 2-4, means that for each KSI of tensile stress applied to the material (above the threshold), 3.7 counts per square millimeter are produced. For tests which involve numerous specimens, relative cracking damage may be assessed by comparing the individual line slopes.

The data analysis techniques described here were applied to all Lockheed test specimens in order to assess differences which may exist between supersonic tunnel and ordinary laboratory oven tests, and the results of this detailed study are presented in a later section of this report.

NASA Specimen Configuration, Preparation, and Analysis

The use of the NASA configuration self-stressed bend specimen offers a number of distinct advantages for hot salt stress corrosion tests. Among them are low specimen preparation costs, elimination of a need for specimen holding fixtures and minimal space requirements due to their small physical size. In addition the amount of stress corrosion damage on these specimens can be ascertained quickly and easily by means of a simple room temperature compression test. The details of the NASA specimen dimensions, construction and method of analysis are adequately reported elsewhere and will not be recapitulated here. For this information the reader is referred to Reference 2.

The NASA specimens were salt coated by spraying on a saturated salt water solution and warming the specimens slightly to promote evaporation of the excess moisture. Successive coatings were applied until a salt thickness of between 0.001 and 0.002 inches was attained.

*It has not been proven from a theoretical standpoint why this should be a linear relationship, but it has been empirically determined that a straight line generally provides a reasonably good fit to these data.

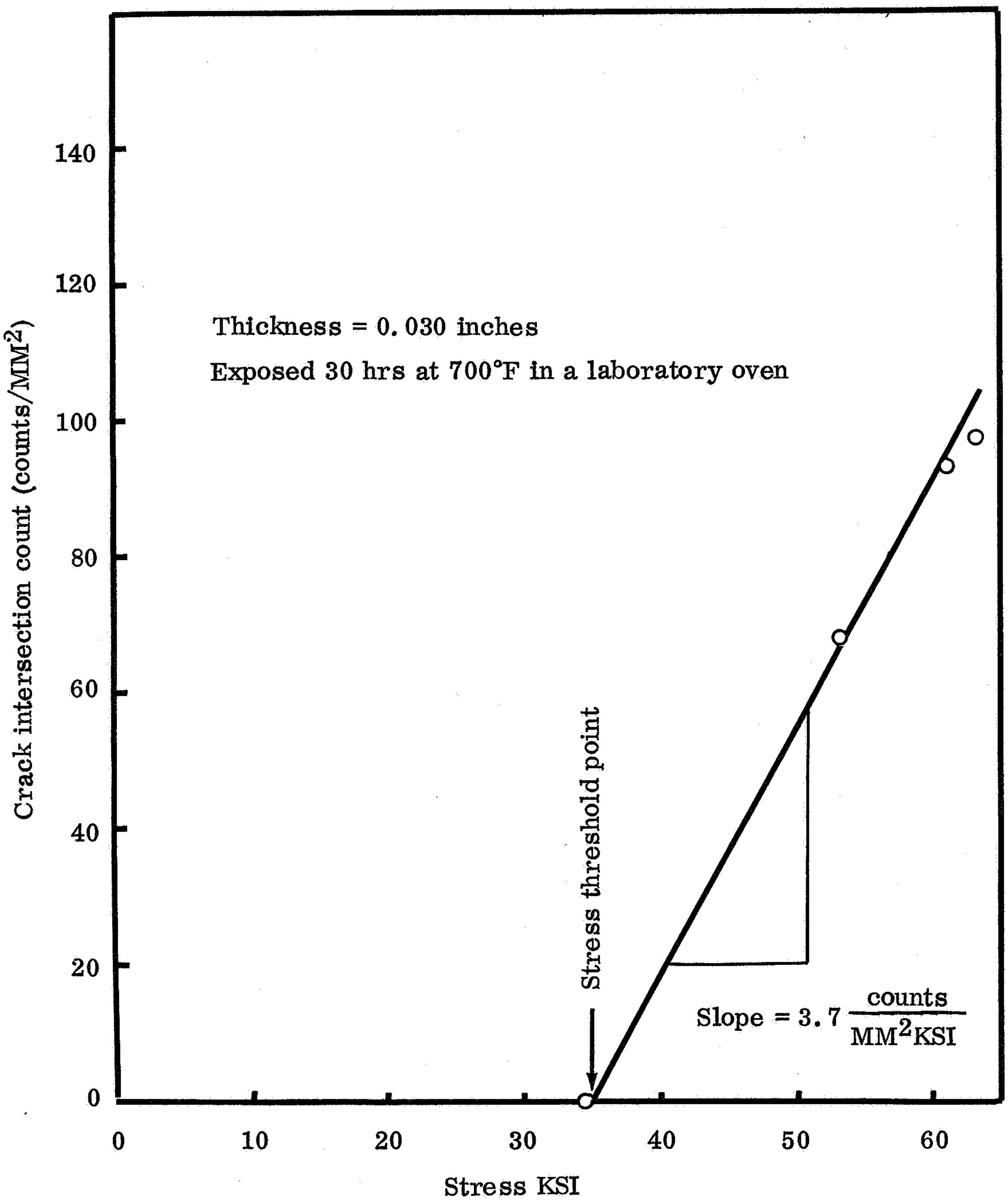


Figure 2-4 CRACK INTERSECTION COUNT AS A FUNCTION OF STRESS LEVEL FOR A TYPICAL Ti-6Al-4V ALLOY LOCKHEED BEND SPECIMEN

Section 3

TUNNEL TEST FACILITY

Description of Tunnel Equipment and Capabilities

In the past, there has been a major problem associated with corrosion testing in supersonic tunnels. In general, stress corrosion experimental procedures require relatively long exposure time to produce observable results. However, most existing tunnel facilities only maintain testing conditions for periods of time which are not of sufficient duration to produce significant corrosion or cracking. A new supersonic tunnel facility at Lockheed (referred to as the Propulsion Tunnel) has eliminated this problem. This test facility enables supersonic air flow conditions to be maintained for indefinite periods of time and is ideally suited for stress corrosion studies. Specimen temperatures in excess of 700°F are obtainable with Mach 2.5 air flow and an ambient pressure level corresponding to an altitude of 70 000 ft.

The Propulsion Tunnel is a part of the Fluid Dynamics Research Facility located at the Rye Canyon Research Laboratory, Saugus, California. The facility is a highly integrated complex, consisting of a Supersonic and Hypersonic Wind Tunnel, a Hypervelocity Shock Tunnel as well as the Propulsion Tunnel.

An air supply-exhaust system serves all the above tunnels, either partially or entirely. Clean, dry air with a minimum dewpoint of -40°F is provided at a rate of 22 lb/sec at 600 psia. Eight tanks with a capacity of 144 000 pounds of air comprise the central air storage plant. A gas fired heater, rated at 14 000 000 Btu/hr. delivers 6 lb/sec of 1200°F air on a continuous basis. The heated air is distributed to the various users by a five-inch diameter manifold.

The facility exhauster plant consists of three centrifugal machines with maximum inlet volume flows of 115 000 cfm, 80 000 cfm, and 20 000 cfm. A vacuum manifold connects the various tunnels to the exhauster plant. The maximum exhauster air flow rate is 214 lb/sec at sea level and the system blank-off is in excess of 150 000 ft pressure altitude conditions.

The Propulsion Tunnel is a versatile test chamber which permits either aerodynamic or propulsion studies. The basic tunnel testing area is a rectangular box, 7 feet high, 6 feet wide, and 22 feet long. A transition section mates the rectangular test section with a 6 foot diameter cylindrical section, providing an overall length of 52 feet (Fig. 3-1). Fourteen 18 inch diameter window openings and ten 8 inch diameter instrumentation penetrations are provided. In addition, 30 inch by 60 inch access doors are provided in either side of the rectangular section.

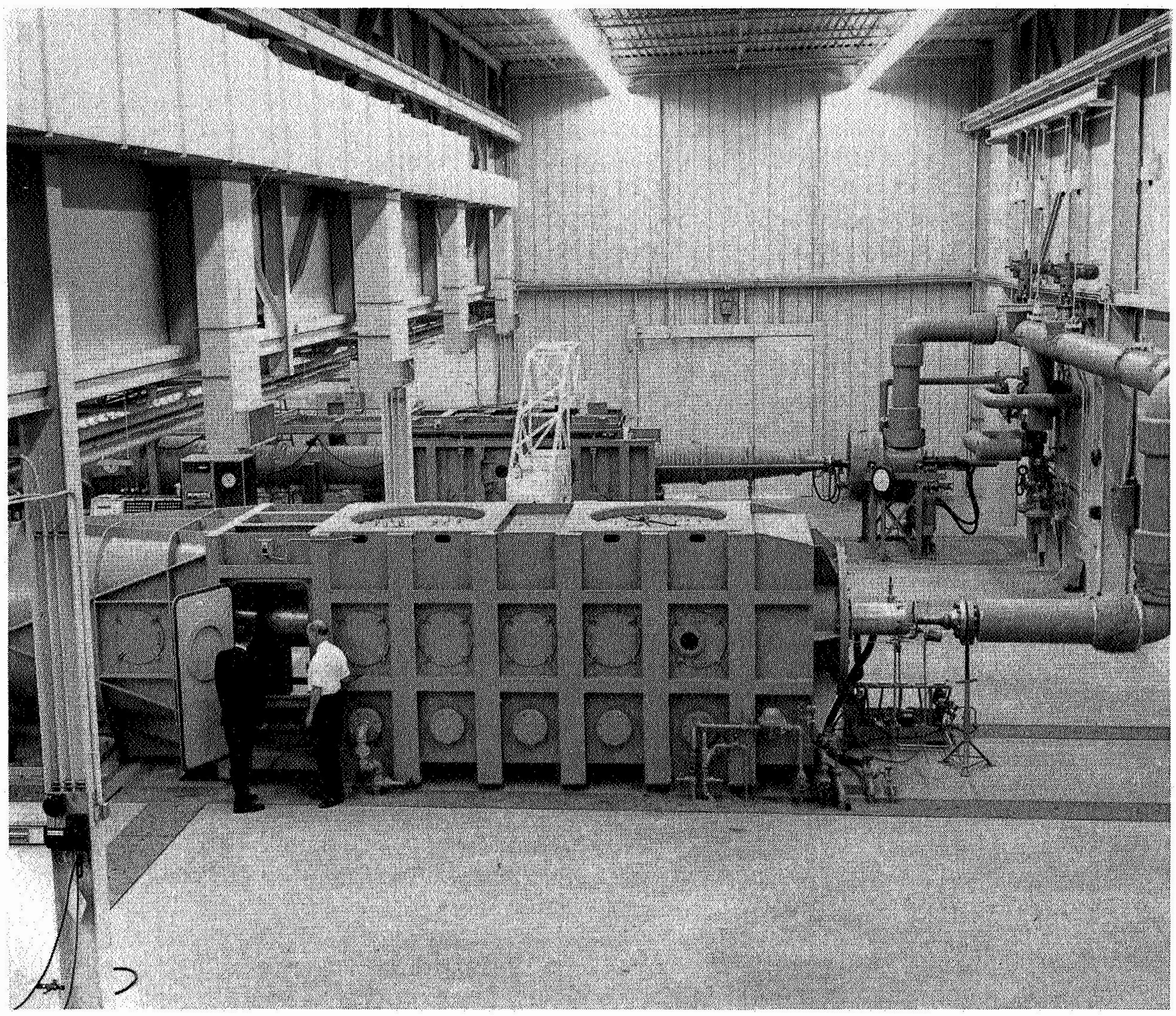


Figure 3-1 LOCKHEED'S SUPERSONIC PROPULSION TUNNEL

Control of air flow to the Propulsion Tunnel is accomplished by means of a thermal mixing valve and 6 inch and 2 inch balanced control valves in parallel. These valves may be regulated by either an electronic controller or a manual hand wheel. Water spray coolers are available to reduce the exit air temperature to a level acceptable to the exhauster plant. Any cooling water which may be carried over into the exhausters is self-drained by means of a barometric well.

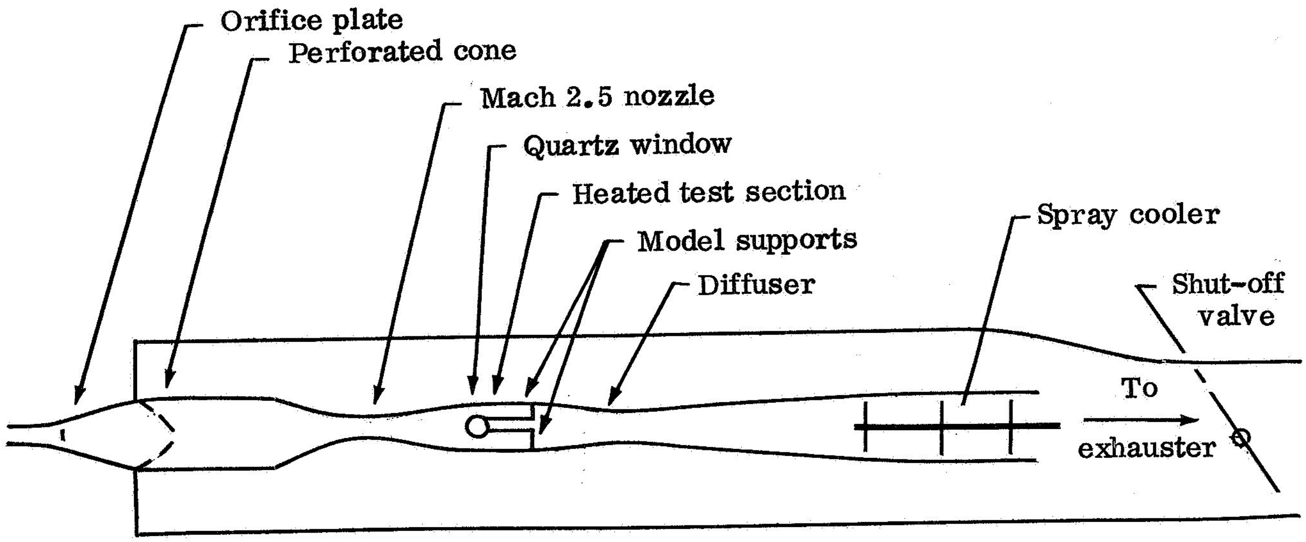
For this test program, the Propulsion Tunnel was set up as a Mach 2.5 wind tunnel. The tunnel components consist of a 20 inch diameter stilling chamber, an axisymmetric 13.23 inch diameter Mach 2.5 nozzle, a closed constant area test section 26 inches in length and a converging-diverging diffuser which expands into a 24 inch diameter constant area duct. The duct contains five cruciform water spray bars. Two 8 inch diameter quartz view ports are provided in the exterior side walls. The windows are connected with the test section by means of extension tubes (Refer Fig. 3-2).

The test section was flanged 12 inches behind the nozzle exit to allow access to the test specimens. The test section diffuser and spray chamber were track mounted and a three foot downstream movement was possible. To protect the outer Propulsion Tunnel shell during prolonged high temperature periods, the entire inner ducting system was insulated. Three inches of fiberglass wrapped with asbestos cloth were applied to all areas except the electrically heated test section. This section was insulated with a three inch thickness of magnesia cement (Refer Fig. 3-3).

In case of an emergency shutdown during the tunnel test, provision was made to ensure that the specimens would not undergo a large temperature drop. This was accomplished by installing twelve heating elements on the test section wall. A total heating capacity of 22.2 kw was utilized. A heat transfer calculation indicated that these heaters would maintain the specimens at 600°F or above during shutdown periods.

Installation and Instrumentation of Tunnel Test Specimens

The individual tunnel test specimens were fastened to aerodynamically configured mounting plates in the tunnel test section. Both Lockheed and NASA mounting plates are illustrated in Fig. 3-4 with the tunnel test section wall cut away to allow visibility. On each of these mounting plates there is space for 32 specimens. In the case of the Lockheed mounting plate, half of the 32 were fastened to one side of the plate and the remainder to the other side. Two slightly different heights were used for the Lockheed specimens due to inherent differences in the elastic moduli of the 8-1-1 and 6-4 titanium alloys. Specimens of the 8-1-1 and 6-4 materials were alternated so that every other one was composed of the same material.



(Not to scale)

Figure 3.2 TEST ARRANGEMENT IN PROPULSION TUNNEL

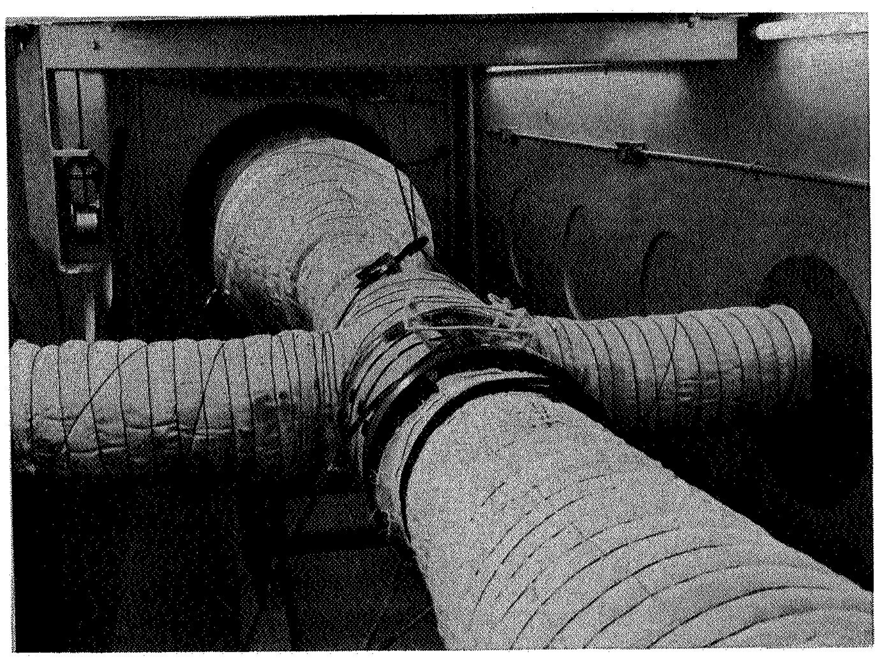


Figure 3-3 INTERIOR VIEW OF PROPULSION TUNNEL SHOWING MACH 2.5 TEST ASSEMBLY

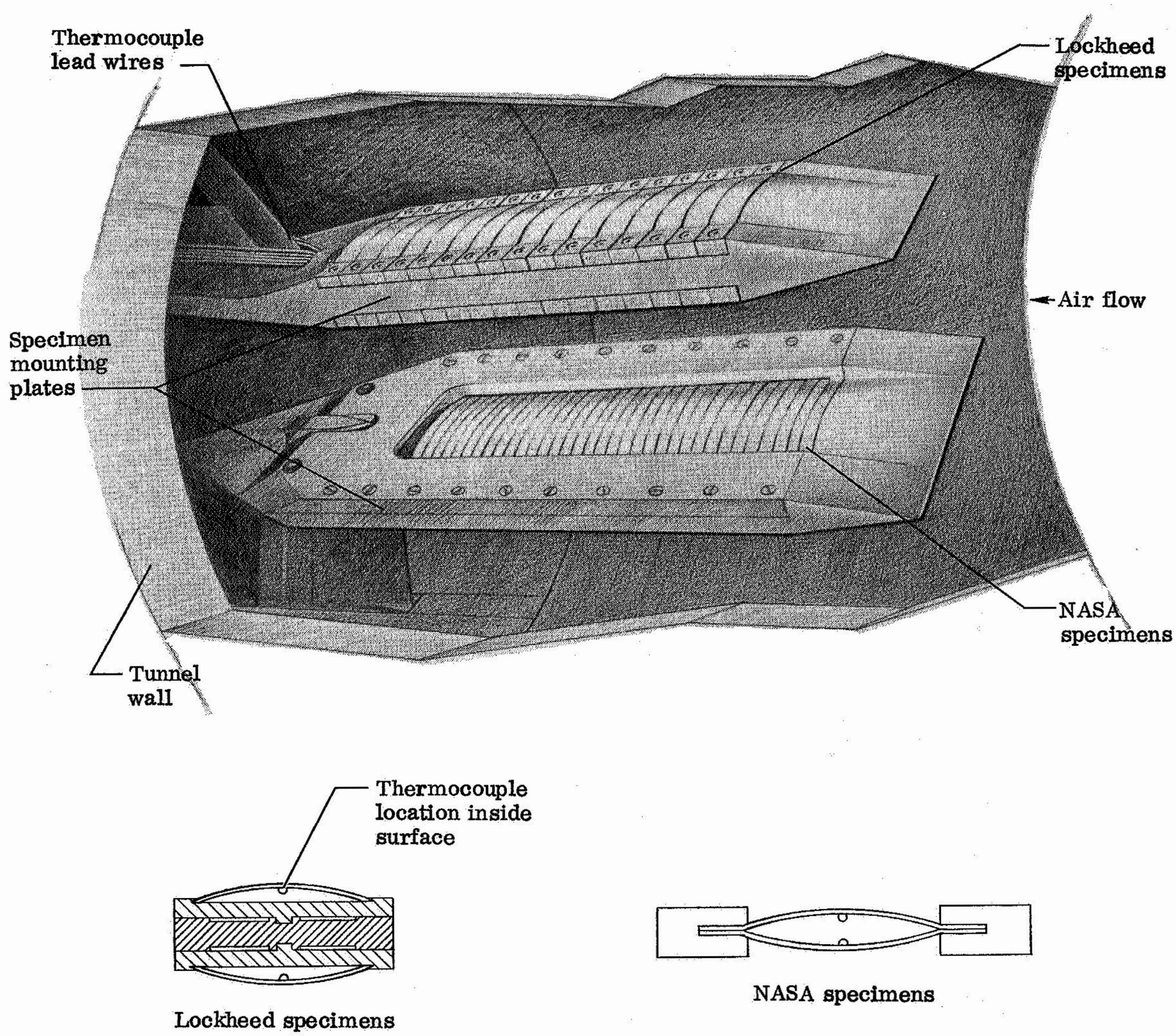


Figure 3-4 ILLUSTRATION OF SPECIMEN MOUNTING PLATES IN TUNNEL TEST SECTION. (A cut-away view of the tunnel wall is shown to allow visibility of the plates.)

For the NASA mounting plate, 32 NASA specimens were placed side-by-side alternating each alloy. Each NASA specimen has two curved surfaces; one curved surface was exposed on one side of the plate and the other was exposed on the other side of the plate.

When place side-by-side, the NASA specimens which were of identical geometry, formed a continuous uniform flow surface. The test surface is not a flat plate, but has curvature, as examination of Fig. 3-4 will indicate. The forward wedges of the specimen holders were contoured to conform with the specimen shape, thus preventing abrupt cross sectional changes. For the NASA specimen mounting plate, it was necessary to make the holder thicker than the maximum specimen height in order to obtain the necessary structural integrity. As a result, the flow over the specimens was three-dimensional in nature. Each plate was cantilever-mounted from the tunnel side wall. The axial position of the mounting plates in the tunnel was adjustable by rebolting the support strut into a new position, thus the mounting plates were capable of being installed at various viewing angles with respect to the quartz windows. The installed configuration was very similar to the classical Busemann biplane, with the two mounting plate centerlines separated by 3.62 inches. It was originally desired to provide electric heating elements in the mounting plates as well as in the test section shell, however, the necessity of obtaining a minimum blockage area precluded the installation of heating elements in the plates.

In order to monitor temperature, certain representative specimens were instrumented with thermocouples. The thermocouples used were #28 gauge chromel-alumel wire enclosed in a 0.060 inch diameter stainless steel sheath. Ball-type thermocouple junctions formed by a mercury-oil arc method were used (Ref. 3). It was believed that welding or peening of the thermocouple junction to the specimen would be unsatisfactory for this test program. The Lockheed specimen thermocouples were brought through a close fitting hole in the bottom of the holding fixture and forced into physical contact with the specimen. The junction was then potted in place with a silicone rubber adhesive. Installation of the NASA specimen thermocouples was somewhat more difficult. The NASA specimen thermocouple sheaths were silver soldered together to provide mechanical rigidity and prevent the junctions from being separated from the specimens during installation in the tunnel. The junctions were formed in the same manner as discussed above for the Lockheed specimens, and potted into place with the silicone adhesive.

The tunnel stagnation pressure was measured with a strain gauge transducer. The stagnation temperature was measured with an iron-constantan bare junction thermocouple. Both stagnation pressure and temperature were displayed on a digital readout device as pure numbers for the convenience of the tunnel operators. All specimen temperatures were recorded on a strip chart recorder of 0 - 1000°F range.

Flow visualization was accomplished by means of an f10 focal length portable Schlieren system. The 8 inch diameter quartz windows used were not of high optical quality, but were sufficiently uniform to allow direct observation of the supersonic flow patterns in the vicinity of the mounting plate leading edges.

Section 4

TEST PROCEDURE

Preliminary Tunnel Tests

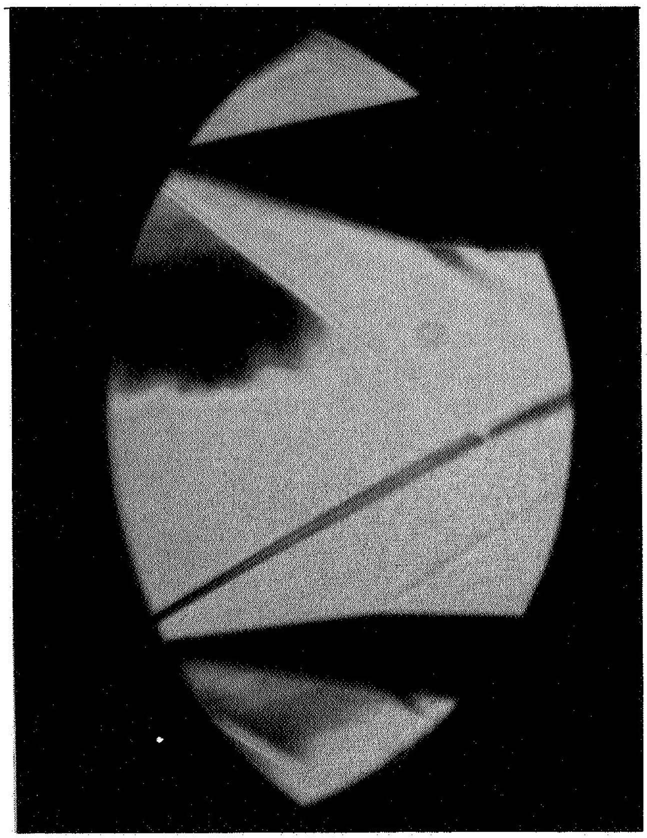
In order to ensure that the tunnel facility would function satisfactorily during the corrosion tests, a relatively short preliminary tunnel test run was made using disposable Lockheed and NASA test specimens. The results of this preliminary run will be briefly described here for completeness.

During the preliminary run alternate top and bottom specimens were instrumented with thermocouples. Space limitations allowed a maximum of 16 thermocouples per holder (a total of 32 couples) with 8 thermocouples distributed on each mounting plate surface. Prior to the preliminary run, the specimens were coated with salt by spraying on a saturated salt solution and allowing the excess moisture to evaporate.

For this preliminary test, the specimen mounting plates were rotated into a horizontal plane to allow Schlieren photography of the flow patterns around the foremost portions of the plates. This optical system rendered the shock wave patterns readily visible, and demonstrated that valid supersonic flow conditions were established. A Schlieren photograph of the flow patterns near the leading edges of the mounting plates is shown in Fig. 4-1.

The temperature data obtained from the preliminary run demonstrated that there was considerable specimen-to-specimen temperature variation, and that the shape of the specimen temperature distribution curve was somewhat dependent on the particular air stagnation temperature employed during the test. Thus, since the final tunnel corrosion tests would not be run under the exact same temperature conditions as the preliminary test, it was deemed necessary to instrument at least some of the final test specimens with thermocouples to ensure a more reliable determination of temperature.

The preliminary tunnel run accomplished its purpose and was considered a major milestone in this program. It demonstrated beyond all doubt that Lockheed's Propulsion Tunnel was capable of operating for indefinite periods of time under pressure conditions corresponding to 70 000 ft altitude, with Mach 2.5 air flow and a specimen temperature in excess of 700°F. The information gained during the preliminary run was an invaluable aid in establishing reliable test facility calibration data and was instrumental in ensuring the success of the critical final test phase of the program since minor operational problems encountered were resolved.



Lockheed specimen mounting plate

NASA specimen mounting plate

Figure 4-1 SCHLIEREN PHOTOGRAPH OF SUPERSONIC FLOW PATTERNS NEAR THE LEADING EDGES OF THE SPECIMEN MOUNTING PLATES

Final Tunnel Tests

Tunnel preparation and thermocouple installation. - After completing the preliminary tunnel tests, the preliminary specimens were removed and the mounting plate assembly was rotated 30° to the horizontal. Tilting in this manner allowed direct visual access (through the quartz viewing windows) to the surfaces of about half of the tunnel specimens.

All tunnel specimen manipulation and installation were performed with great care by a highly skilled instrumentation specialist. When handling the test specimens, caution was exercised to prevent contamination and ensure that the salt layer on the precoated NASA specimens would not be damaged. Disposable polyethylene clean gloves were worn during the set-up phase of the work, and whenever possible, physical contact was restricted to the very ends or the undersides of the specimens. A total of 16 specimens were instrumented with sheathed chromel-alumel thermocouples using the techniques described earlier (Refer to Section 3 for details). Four additional couples were securely fastened to the aft ends of the specimen mounting plates. To determine the placement of these various couples on the plates refer to Figs. 4-2 and 4-3. There were four instrumented specimens on both surfaces of each mounting plate. The possibility of using more couples was considered, but decided against since additional instrumentation work was severely hampered by space limitations and would have required excessive manipulation of the specimens, increasing the risk of damage or contamination.

Prior to testing, the precoated NASA specimens were protected with a polyethylene sleeve which fit over the NASA mounting plate. Then the Lockheed specimens were coated in situ using the techniques described in Section 2. In order to hasten the evaporation process a stream of clean warm air was directed over the specimens. Coating was continued until an approximate salt thickness of about 0.003 inches was attained.

Specimen position and exposure. - The actual tunnel exposure phase of the program consisted of three separate tunnel runs (hereafter referred as runs I, II, and III). Runs I and II were each of 15 hours duration with an average equilibrium specimen temperature being attained of just over 700°F. Run III was 20 hours in length with an average specimen temperature slightly in excess of 600°F. These temperatures were selected by joint agreement between the Lockheed and NASA personnel. In all three runs, Mach 2.5 air flow and 70 000 ft pressure altitude conditions were maintained.

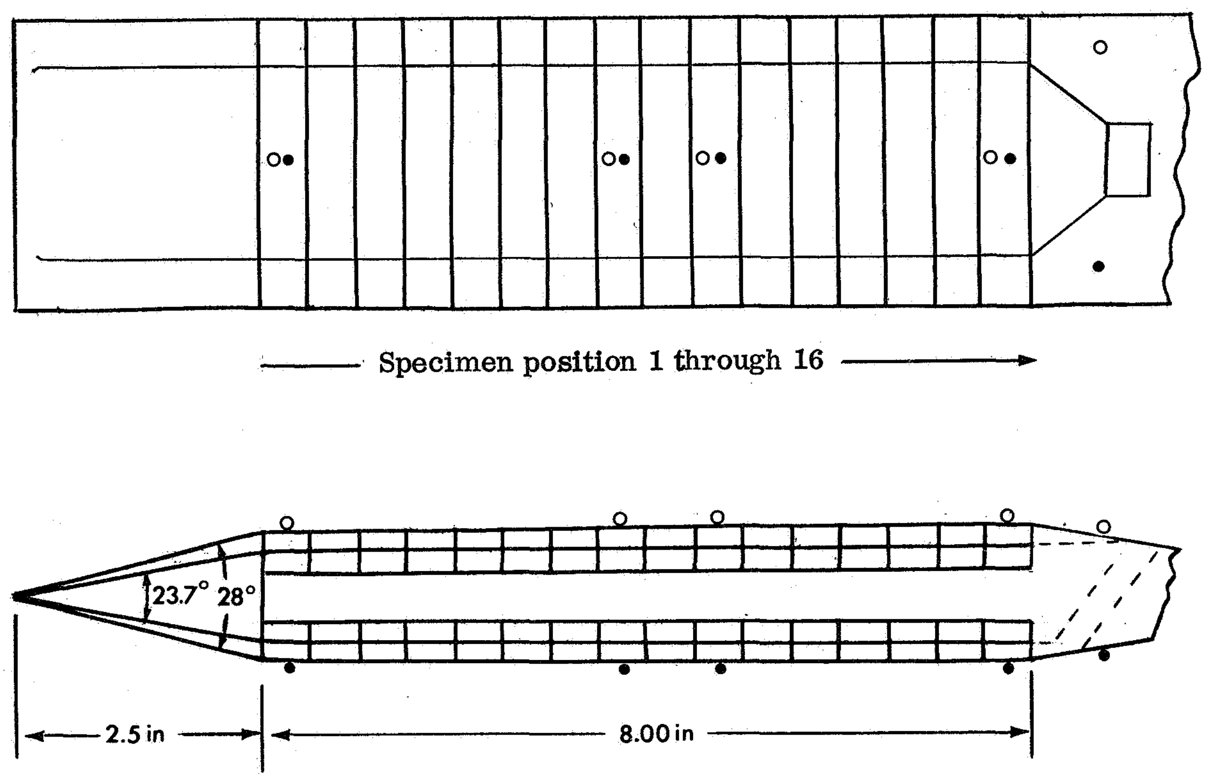
After completion of Runs I and II, specimens were removed from the tunnel test section and replaced with new ones to provide varying exposure experiences. Table 4-1 presents information relative to the exposure of the various specimens. A total of 32 specimens were exposed at 700°F for 15 hours during Runs I and II; 32 specimens were exposed at 600°F for 20 hours during Run III, 16 specimens at 700°F for two 15 hour cycles (Runs I and II), and 32 specimens at 700°F for two 15 hour cycles and 600°F for a 20 hour cycle (Runs I, II, and III).

After Run I it became necessary to remove all the NASA specimen thermocouples and all but four of the Lockheed specimen thermocouples. The necessity for removal of the NASA couples was due to the fact that all the lead wires were arranged to run back through the centers of the specimens. Thus, in order to remove NASA specimens near the rear of the mounting plate, all the lead wires had to be disconnected. However, for the Lockheed specimens, it was possible to run the wires beneath the individual stainless steel specimen holders, hence it was only necessary to disconnect couples when instrumented specimens were replaced. After Run II, specimen exchange required the removal of all but one Lockheed specimen thermocouple. Fortunately, the four couples on the aft end of the mounting plates remained intact throughout the entire 50 hour tunnel test period.

In order to supplement the thermocouple data, a remote radiometer was used to determine the temperatures of the specimens which were visible through the quartz viewing windows. Calibration of the radiometer was accomplished during Run I by adjusting the radiometer readings to coincide with thermocouple

○ – Denotes thermocouples on outer side of plate

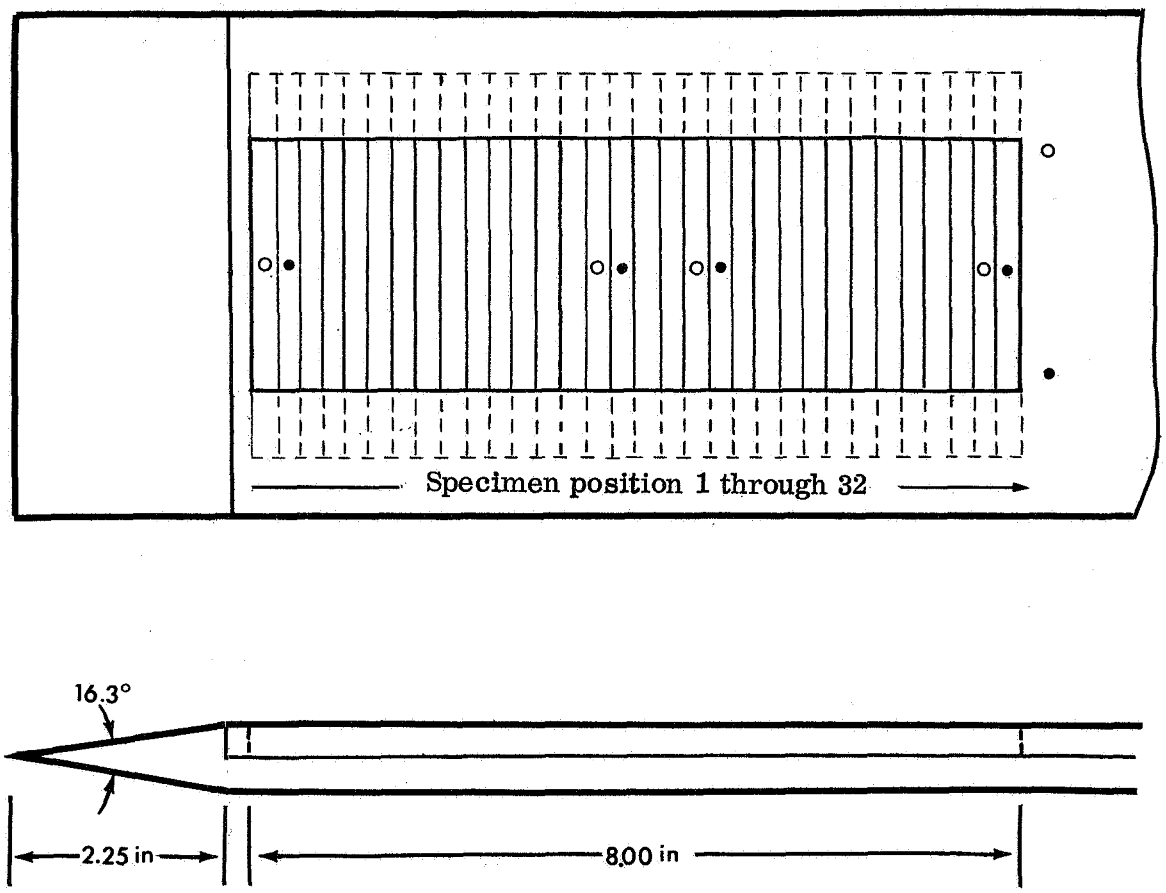
• – Denotes thermocouples on inner side of plate



Note: All thermocouples were located inside (compressively stressed surface) of the specimens.

Figure 4-2 TOP AND SIDE VIEWS OF LOCKHEED SPECIMEN MOUNTING PLATE SHOWING PLACEMENT OF THERMOCOUPLES AT THE BEGINNING OF THE TUNNEL TEST o – Denotes thermocouples on outer side of plate

• – Denotes thermocouples on inner side of plate



Note: All thermocouples were located inside (compressively stressed surface) of the specimens.

Figure 4-3 TOP AND SIDE VIEWS OF NASA SPECIMEN MOUNTING PLATE SHOWING PLACEMENT OF THERMOCOUPLES AT THE BEGINNING OF THE TUNNEL TEST

TABLE 4-1

SPECIMEN POSITION AND EXPOSURE DATA

Run	Nominal temp. °F	Time hrs	Specimen Type	Position from front		Material	Spec. no.	Total spec.
I	700	15	Lockheed	Outer and inner	1	6-4		8
					2	8-1-1		
					15	6-4		
					16	8-1-1		
			NASA	1		6-4	25	8
				2		8-1-1	81	
				3		6-4	53	
				4		8-1-1	29	
				29		6-4	19	
				30		8-1-1	87	
				31		6-4	16	
				32		8-1-1	11	
II	700	15	Lockheed	Outer and inner	1	6-4		8
					2	8-1-1		
					15	6-4		
					16	8-1-1		
			NASA	1		6-4	29	8
				2		8-1-1	23	
				3		6-4	45	
				4		8-1-1	43	
				29		6-4	8	
				30		8-1-1	37	
				31		6-4	18	
				32		8-1-1	19	

TABLE 4-1. - Continued

SPECIMEN POSITION AND EXPOSURE DATA

Run	Nominal temp. °F	Time hrs	Specimen Type	Position from front		Material	Spec. no.	Total spec.
III	600	20	Lockheed	Outer and inner	1	6-4		16
					2	8-1-1		
					3	6-4		
					4	8-1-1		
					13	6-4		
					14	8-1-1		
					15	6-4		
					16	8-1-1		
			NASA	1		6-4	26	16
				2		8-1-1	93	
				3		6-4	79	
				4		8-1-1	84	
				5		6-4	48	
				6		8-1-1	94	
				7		6-4	9	
				8		8-1-1	5	
				25		6-4	11	
				26		8-1-1	85	
				27		6-4	12	
				28		8-1-1	1	
				29		6-4	6	
				30		8-1-1	8	
				31		6-4	13	
				32		8-1-1	98	
I + II	700 700	15 15 30 Total	Lockheed	Outer and inner	3	6-4		8
					4	8-1-1		
					13	6-4		
					14	8-1-1		
			NASA	5		6-4	7	8
				6		8-1-1	46	
				7		6-4	54	
				8		8-1-1	55	
				25		6-4	43	
				26		8-1-1	97	
				27		6-4	27	
				28		8-1-1	28	

TABLE 4-1. - Continued

SPECIMEN POSITION AND EXPOSURE DATA

Run	Nominal temp °F	Time hrs	Specimen Type	Position from front	Material	Spec. no.	Total spec.
I + II + III	700, 700, 600	15, 15, 20; 50 Total	Lockheed	Outer and inner 5	6-4		16
				6	8-1-1		
				7	6-4		
				8	8-1-1		
				9	6-4		
				10	8-1-1		
				11	6-4		
				12	8-1-1		
			NASA	9	6-4	32	16
				10	8-1-1	32	
				11	6-4	28	
				12	8-1-1	82	
				13	6-4	38	
				14	8-1-1	21	
				15	6-4	72	
				16	8-1-1	9	
				17	6-4	10	
				18	8-1-1	27	
				19	6-4	44	
				20	8-1-1	70	
				21	6-4	50	
				22	8-1-1	14	
				23	6-4	15	
				24	8-1-1	62	

temperatures on instrumented specimens. Calibration of the radiometer revealed that the thermal emissivity value of the coated specimens was surprisingly low (approximately 0.42 for Lockheed and 0.45 for NASA specimens). This low value is characteristic of metal surfaces, indicating that the salt was acting as a partially transparent media, allowing the metal surface below to be "seen" by the radiometer. The slightly lower average emissivity value for the Lockheed specimens may be due to the highly polished surface. In general radiometric measurements are not as accurate as thermocouple measurements. Erroneous readings on radiometers may arise from reflections, surface roughness, oxide formation, etc. However, in those cases when little or no thermocouple data were available, the radiometric data aided in predicting the temperature. By combining the available thermocouple data with the radiometer data and interpreting these data in terms of test results for previous runs, it was possible to estimate temperature conditions for all the tunnel specimens. The thermocouples at the aft end of the mounting plate proved to be a valuable aid in predicting the specimen temperatures, particularly after equilibrium conditions were achieved.

Transient thermal response. - The transient thermal response curve for each tunnel run was established by determining the average specimen temperature at each increment of time and plotting these data as a function of the exposure time (Figs. 4-4 through 4-6). Inspection of these curves revealed that periods of time between about 30 and 50 minutes were required to establish equilibrium conditions for the various tests. Notice that for each tunnel run the initial slopes of the heating curves are somewhat steeper for the NASA than for Lockheed specimens. This variation in heating rate may be due in part to differences in the thermal capacities of the two types of specimens. The large thermal mass associated with the Lockheed specimen holder might be expected to result in a somewhat lower heating rate. In addition, part of these heating rate differences may be due to the fact that the aerodynamic configurations and the flow characteristics are not identical for the two plates; thus the heat transfer rates were not completely uniform for all specimens.

Notice that there are some differences in the initial shapes of the Lockheed specimen transient response curves for Runs I and II even though the test conditions in these two cases were essentially identical. These differences are not deemed particularly significant, since in the first case the curve was based on the average data of eight thermocouples, and in the second case the curve was established from only four couples, since some had been removed. Thus, the temperature data shown in these two cases probably reflect slight differences in heating rates of specimens at different geometric locations on the mounting plate.

The only technical difficulty encountered during these tests is reflected in the temperature data from Run II after 682 minutes exposure (11.3 hours). At this point a malfunction in the air heater occurred, requiring an emergency tunnel shutdown. The heaters in the test section shell were activated, preventing the average specimen temperature from dropping below 650°F. The trouble was quickly eliminated and the tunnel was reactivated within eight minutes after shutdown. In view of the fact that the shutdown time was short

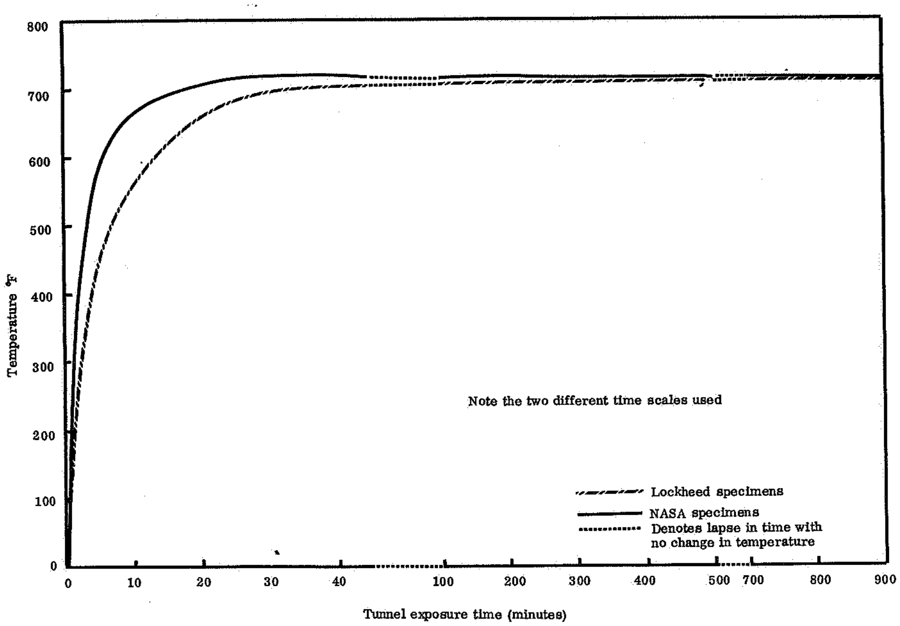


Figure 4-4 AVERAGE SPECIMEN TEMPERATURE FOR THE FIRST 15 HOUR TUNNEL TEST (TUNNEL RUN I)

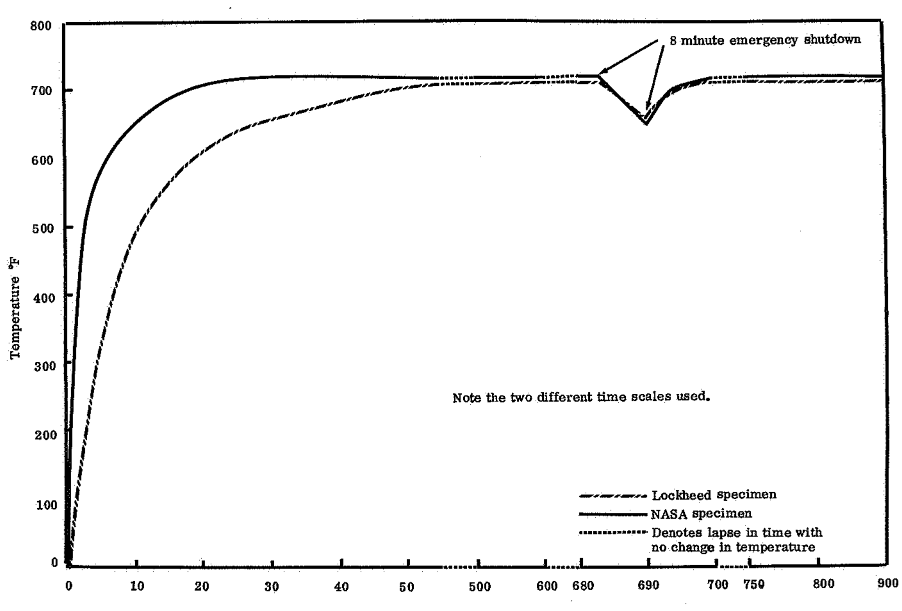


Figure 4-5 AVERAGE SPECIMEN TEMPERATURE FOR THE SECOND 15 HOUR TUNNEL TEST (TUNNEL RUN II)

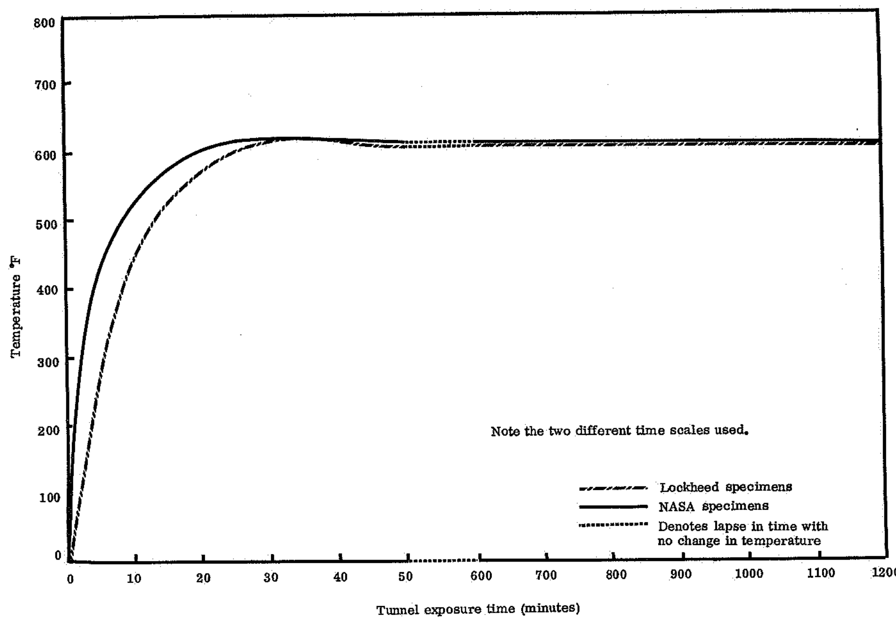


Figure 4-6 AVERAGE SPECIMEN TEMPERATURE FOR THE 20 HOUR FINAL TUNNEL TEST (TUNNEL RUN III)

and the samples did not experience large temperature excursions, it is believed that this malfunction did not appreciably affect the results of these tests.

Specimen Temperature Distributions

The average equilibrium temperature distribution curves for all tunnel test specimens were computed and are shown in Figs. 4-7 through 4-12. When preparing these curves, close inspection of the thermocouple temperature data revealed that the temperature distributions were somewhat dependent on normal small variations of the operating conditions of the tunnel. Small but unavoidable fluctuations in the tunnel operating conditions produced some fluctuation in the individual specimen temperature curves during the course of these tests. The distributions shown represent the average specimen temperatures once steady state conditions were achieved, as indicated by a condition of temperature constancy in the curves of Figs. 4-4 through 4-6.

Notice for tunnel Runs I and II, a maximum temperature spread of about 37°F was indicated for the Lockheed specimens whereas NASA specimens displayed a maximum spread of about 30°F. In tunnel Run III the spread was about 25°F for the Lockheed and about 20°F for the NASA specimens.

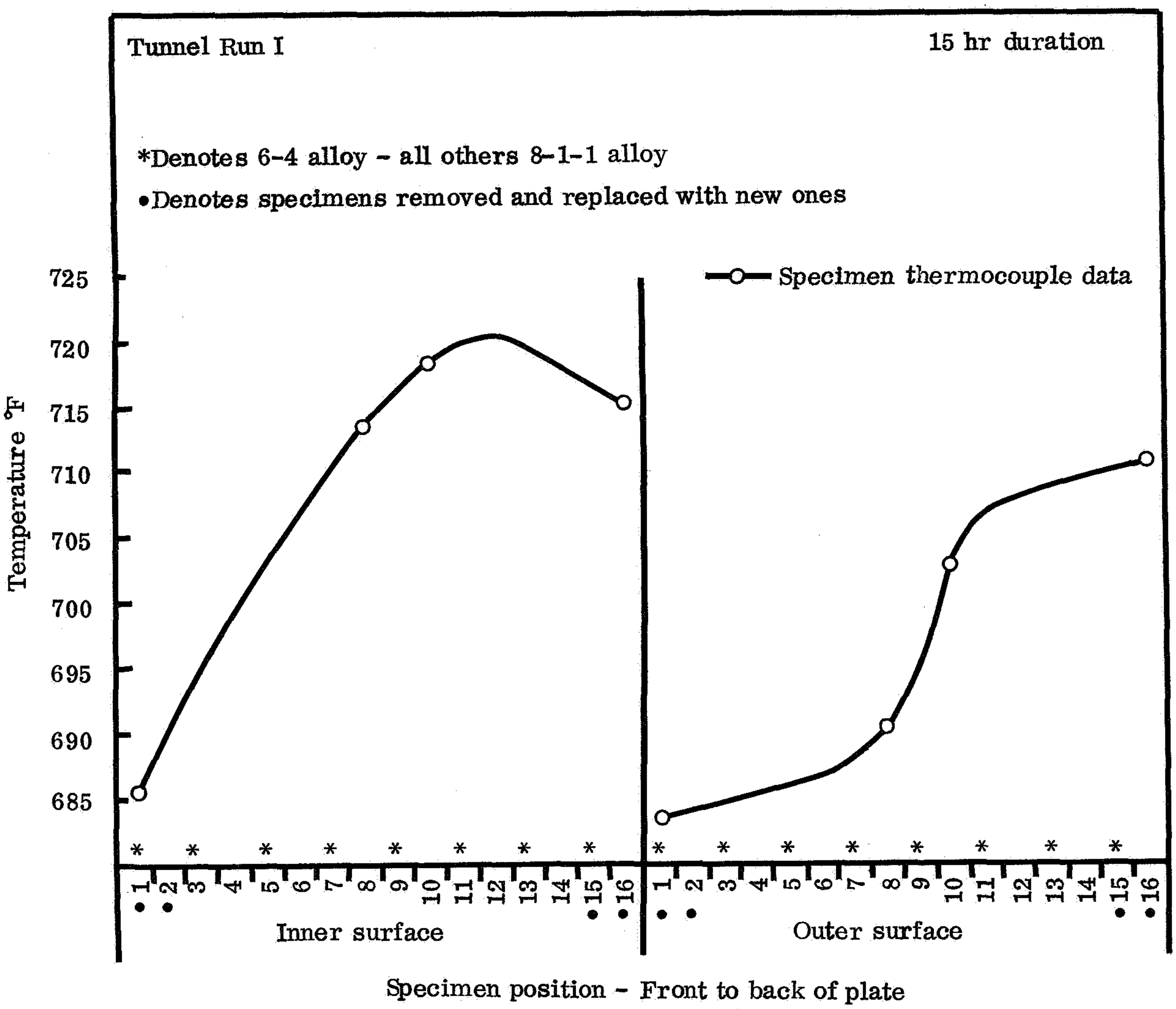


Figure 4-7 AVERAGE EQUILIBRIUM TEMPERATURE OF LOCKHEED SPECIMENS AS A FUNCTION OF POSITION ON TUNNEL MOUNTING PLATE

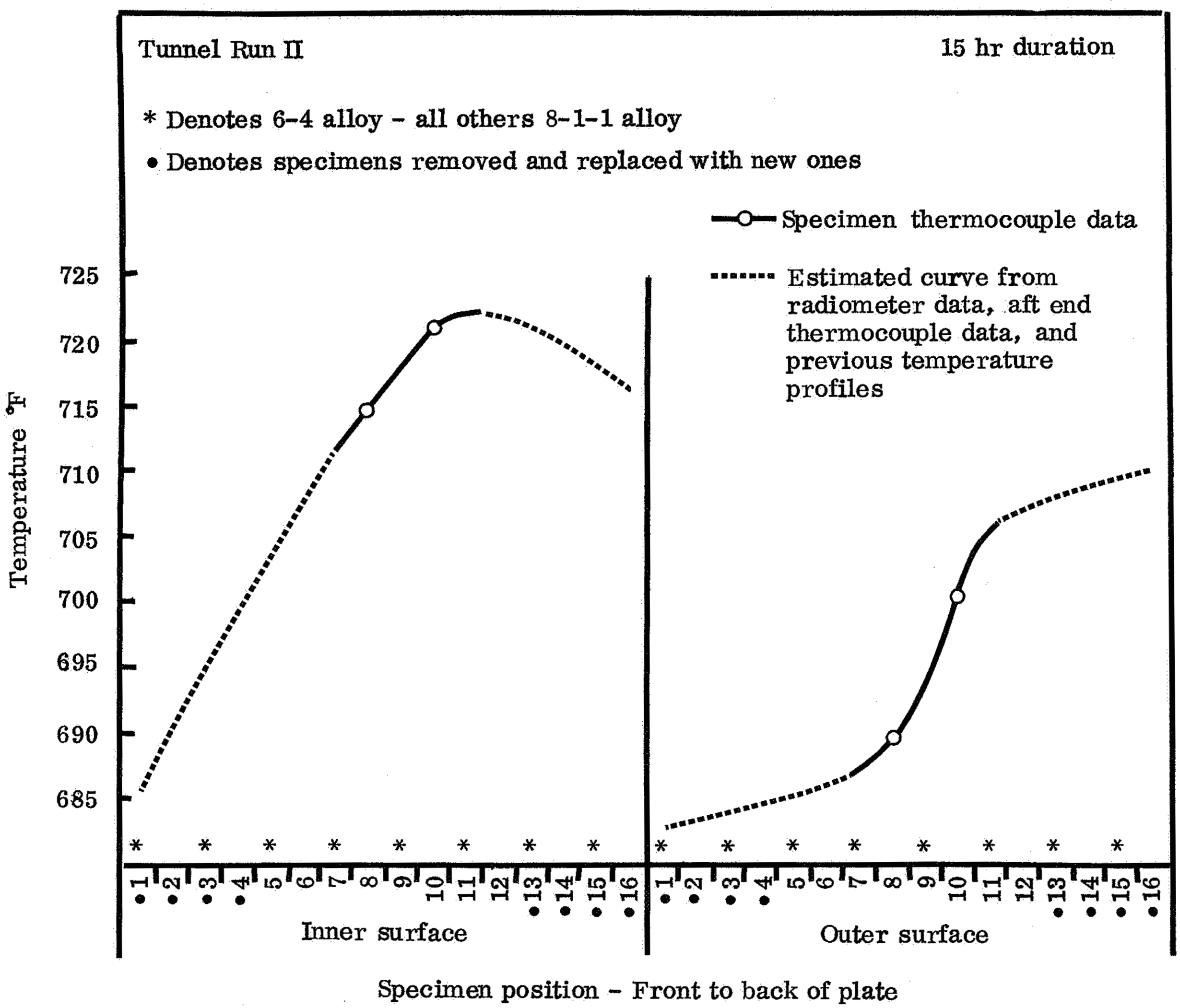


Figure 4-8 AVERAGE EQUILIBRIUM TEMPERATURE OF LOCKHEED SPECIMENS AS A FUNCTION OF POSITION ON TUNNEL MOUNTING PLATE

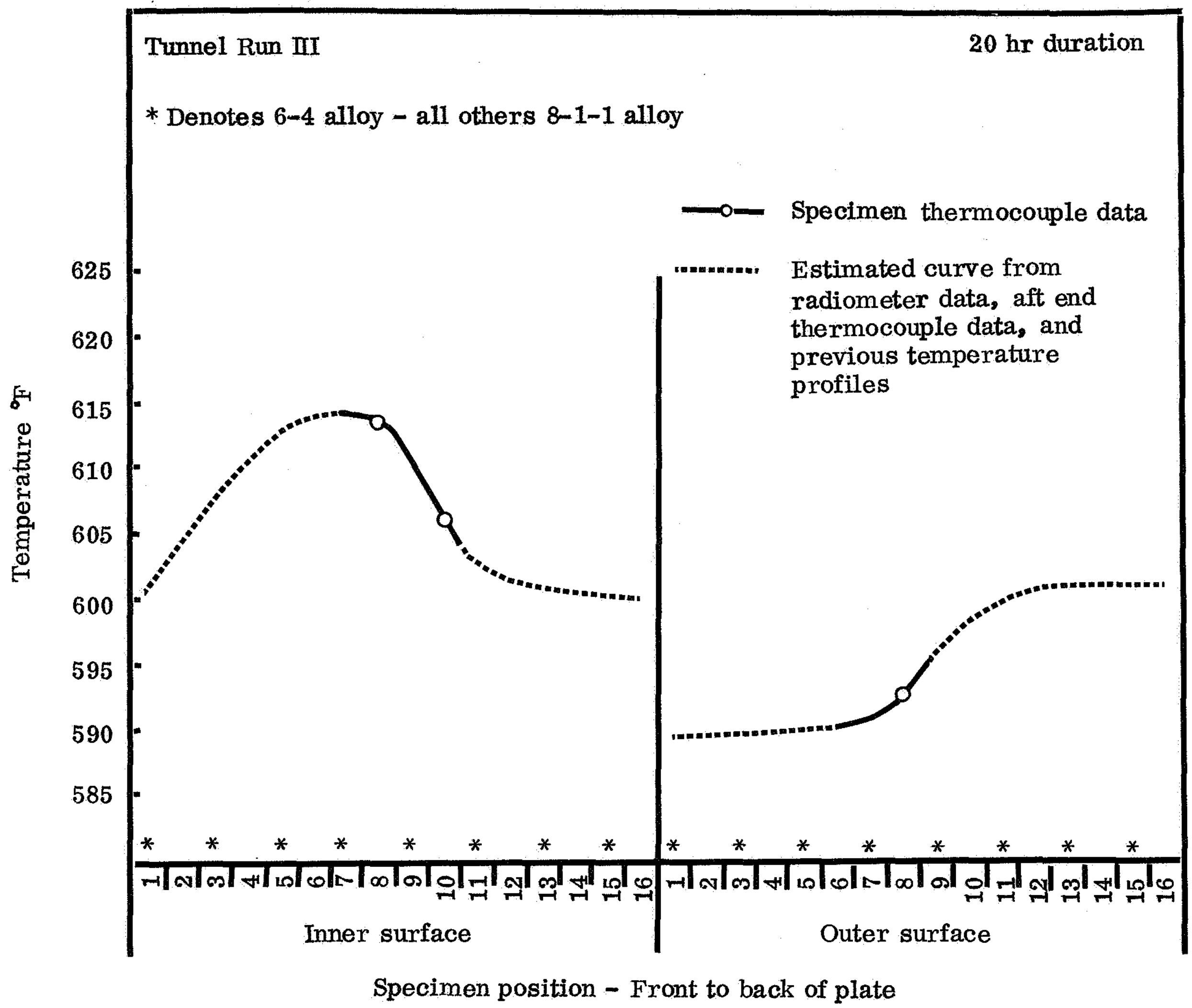


Figure 4-9 AVERAGE EQUILIBRIUM TEMPERATURE OF LOCKHEED SPECIMENS AS A FUNCTION OF POSITION ON TUNNEL MOUNTING PLATE

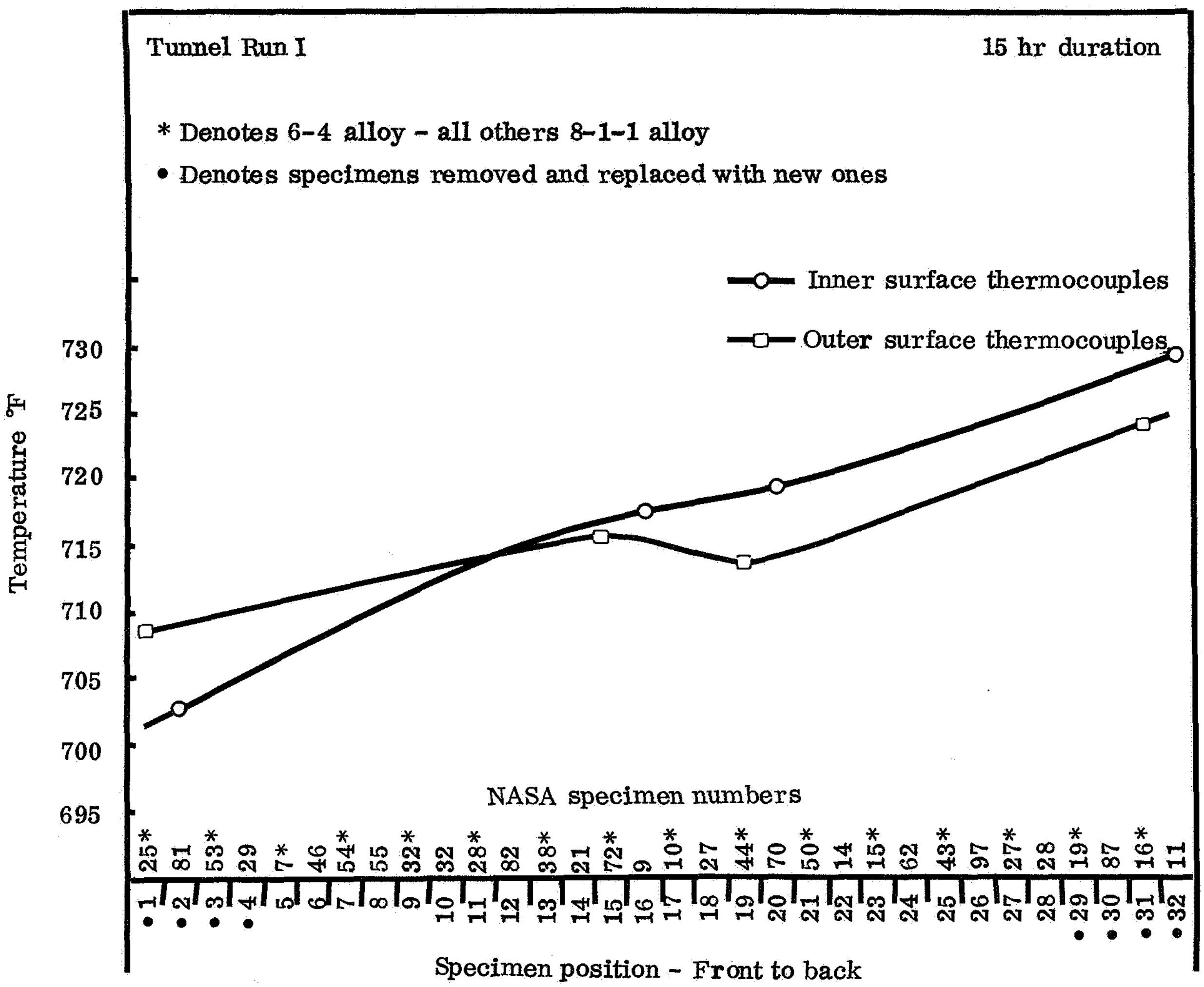


Figure 4-10 AVERAGE EQUILIBRIUM TEMPERATURE OF NASA SPECIMENS AS A FUNCTION OF POSITION ON TUNNEL MOUNTING PLATE

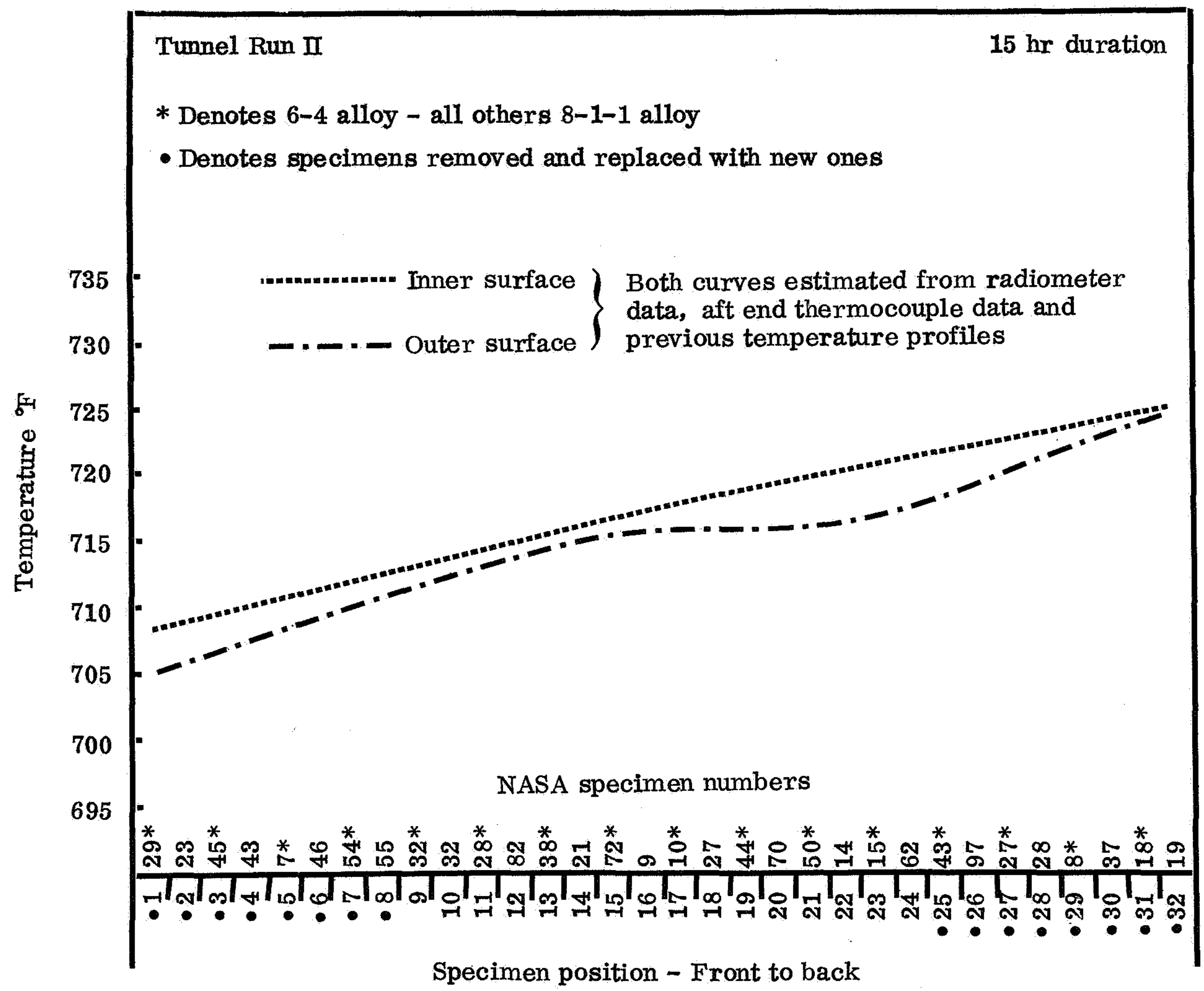


Figure 4-11 AVERAGE EQUILIBRIUM TEMPERATURE OF NASA SPECIMENS AS A FUNCTION OF POSITION ON TUNNEL MOUNTING PLATE

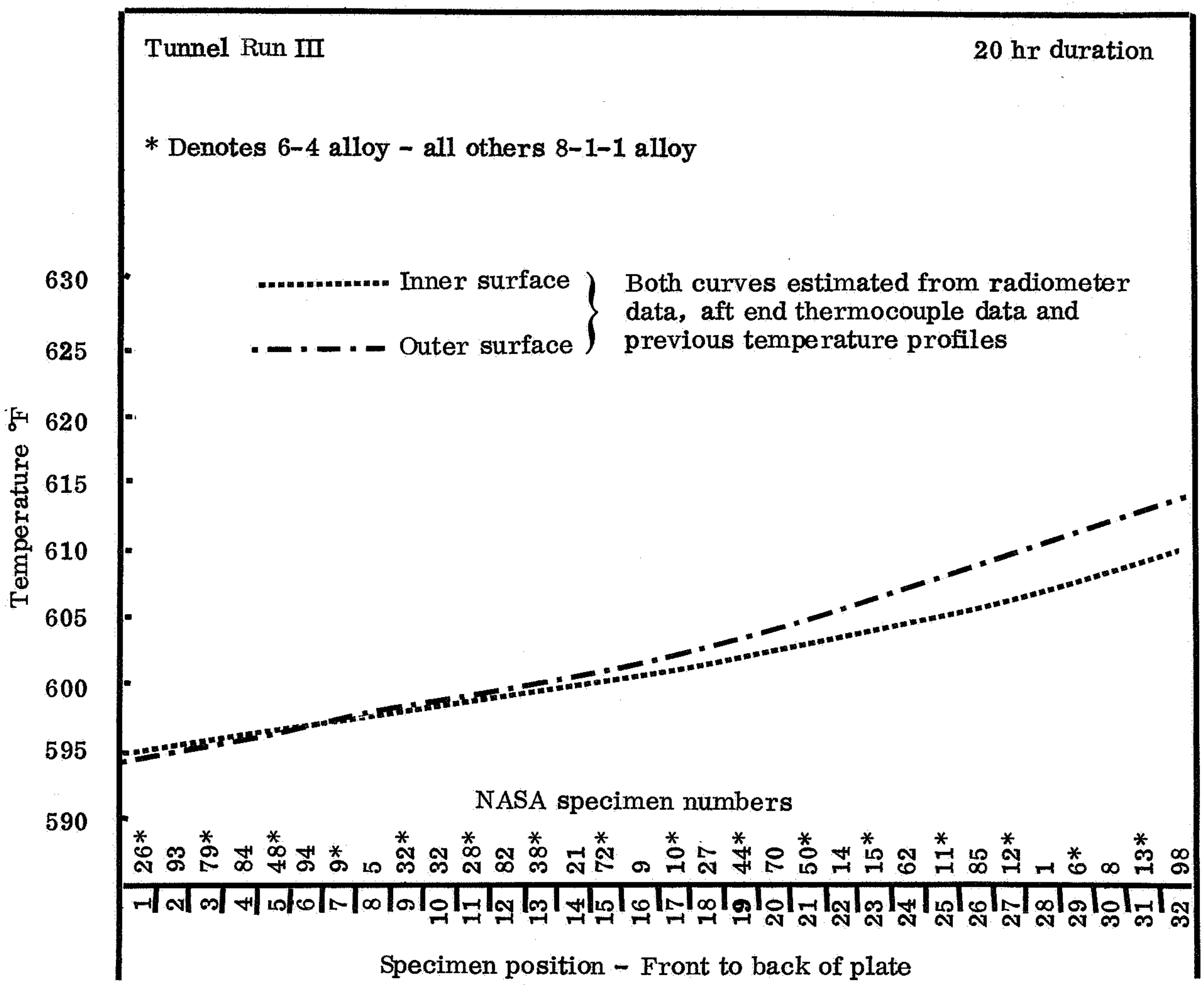


Figure 4-12 AVERAGE EQUILIBRIUM TEMPERATURE OF NASA SPECIMENS AS A FUNCTION OF POSITION ON TUNNEL MOUNTING PLATE

It should be pointed out that average temperature distribution data are probably quite reliable for the first 15 hour run where an adequate number of thermocouples were available, but are not as reliable for later runs in which less thermocouple instrumentation was employed. The temperature distribution data for the later tests must be used only as a guide and should not be regarded as precision data. Portions of curves which were estimated by means other than specimen thermocouples are somewhat questionable and are therefore indicated by dotted lines.

Oven Tests

In order to establish laboratory oven tests which were fairly representative of the temperatures for tunnel Runs I and II, the overall spreads (Figs. 4-7 through 4-12) were approximated by three individual circulating air ovens set at 685, 703, and 717°F. All three ovens were utilized in representing the relatively wide spread of the Lockheed specimens; but only the 685°F and the 703°F oven were used to represent the smaller temperature spread of the NASA specimens. In a similar manner, three ovens were established at 590°F, 595°F and 605°F to cover the spread during the 20 hour run.

The first set of oven tests performed was in direct thermal correspondence to the tunnel tests. In this series of tests, the specimens were divided among the three ovens discussed above. At the beginning of the test the ovens were turned off and established at room temperature conditions. Then the ovens were activated and made to closely follow the transient curves shown in Figs. 4-4 through 4-6 utilizing careful manual control. Each oven approached one of the average equilibirum temperatures mentioned above. As in the tunnel tests, upon completion of a run, specified specimens were removed and replaced with new ones. Even the shutdown times between runs were the same as for the tunnel tests. The eight minute emergency shutdown which occurred during tunnel Run II was included in these oven tests. Thus, for each tunnel Run I, II, and III, there was a corresponding laboratory oven test which was performed using the same number of salt coated samples (112) under identical conditions of exposure temperature and time. Just as in the tunnel tests, specimens with a total accumulated exposure time of 15, 20, 30, and 50 hours were obtained. In order to provide base line data an additional 64 uncoated specimens (16 of each type and alloy) were included in these tests.

In addition to the directly corresponding oven tests, additional exposure tests were performed in ovens set at constant temperatures representative of the average specimen equilibirum temperatures of the tunnel test. For Run I and II the average temperature value was taken simply as the arithmetic mean temperature of the particular ovens used in the directly corresponding tests. For the Lockheed specimens, three ovens were used in the directly corresponding oven test for Runs I and II; the mean temperature of these ovens was 702°F. For the NASA specimens, two ovens were used to represent the temperatures involved in Runs I and II with a mean temperature of 710°F. Constant temperature ovens were established at each of these mean values. For the 20 hour tests, the averages were computed in the same way yielding a mean temperature of 600°F for the NASA and 597°F for the Lockheed specimens. Since these temperatures were so close together a single constant temperature oven at 598°F was utilized.

Thus, using ovens at these three average temperatures, two 15 hour and a final 20 hour exposure test were conducted. At the end of each run all specimens were removed from the oven and allowed to return to room temperature where they remained overnight. Then, as in previous tests, certain specimens were removed and replaced with new ones. The next phase of the test was then initiated by placing the specimens directly into the appropriate constant temperature oven. In these constant temperature oven tests, a total of 64 salt coated specimens was utilized. Of the total 64 used, 16 each were exposed for 15 and 20 hours, 8 were exposed for 30 hours, and 24 for 50 hours. An additional 56 uncoated specimens were included in this test to help establish base line data.

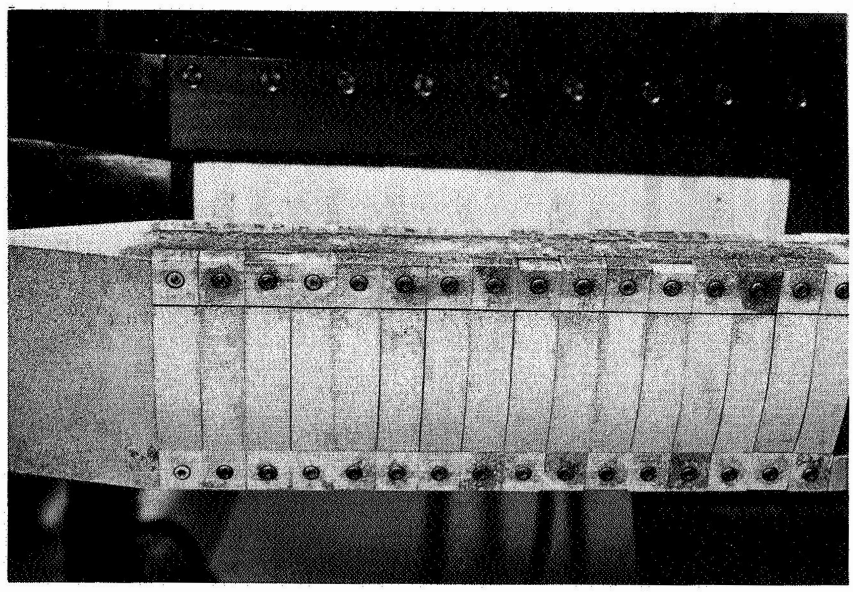
Section 5

RESULTS AND DISCUSSION

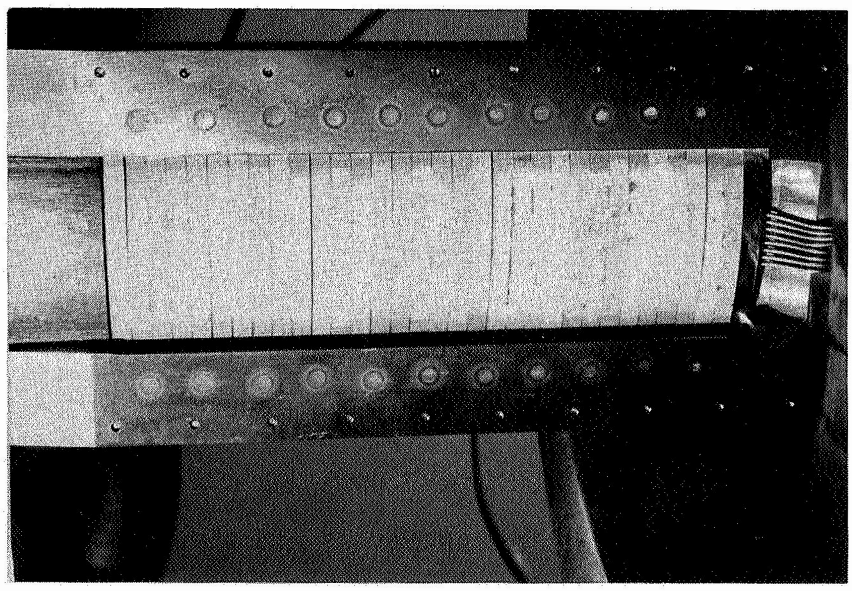
Study of Changes in the Salt Coating

Before being tested, the coated specimens were photographed in place in the tunnel test section. After each tunnel run, the specimens were again photographed to record any visual changes in the salt coating which may have occurred. Due to space limitations, it was impossible to position the camera in such a way to allow photography of all specimens. However, all were photographed with the exception of the Lockheed specimens on the inner surface of the mounting plate Fig. 5-1 shows the specimens in the pretest condition, and Figs. 5-2 through 5-4 shows their appearance after each tunnel run. The specimens indicated with a white oval were removed and replaced with new ones immediately after being photographed. The unmarked specimens remained in place for the next run. By careful inspection one can discern progressive changes in the salt coating of individual specimens.

At the beginning of the tunnel tests all specimens were uniformly coated with salt, and all replacements had adherent continuous coatings. Thus, it is obvious from Fig. 5-2 that a considerable portion of the salt was removed during Run I, and similarly Fig. 5-3 shows that more salt removal occurred during Run II. However, during the final 600°F test (Run III) only a small amount of salt loss occurred (note particularly specimens installed after Run II). This result indicates that the higher temperature conditions characteristic of the first two runs (i.e. 700°F) probably contribute to the loss of adhesion of the salt layer. Visual observation revealed that in each case the majority of the



Lockheed specimens on outer surface of mounting plate



NASA specimens on outer surface of mounting plate

Figure 5-1 APPEARANCE OF SALT COATED TUNNEL SPECIMENS PRIOR TO TESTING

15 hrs duration

Nominal specimen temp. 700°F

NASA inner surface

Lockheed outer surface

NASA outer surface

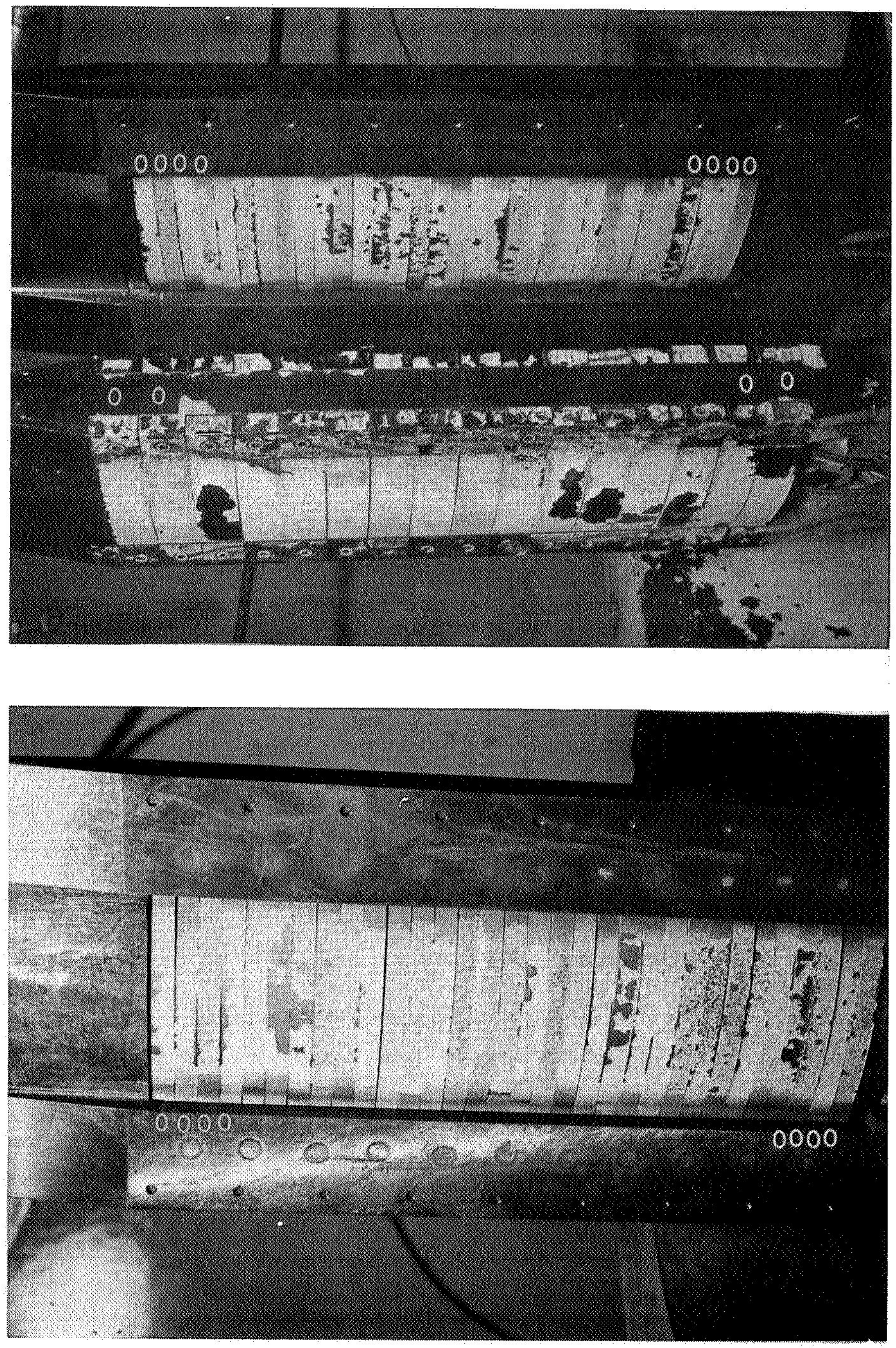


Figure 5-2 APPEARANCE OF TUNNEL SPECIMENS AFTER COMPLETION OF RUN I

NOTE: White oval denotes specimens removed and exchanged with new ones immediately after photograph was taken.

15 hrs duration Nominal specimen temp. 700°F

NASA inner surface

Lockheed outer surface

NASA outer surface

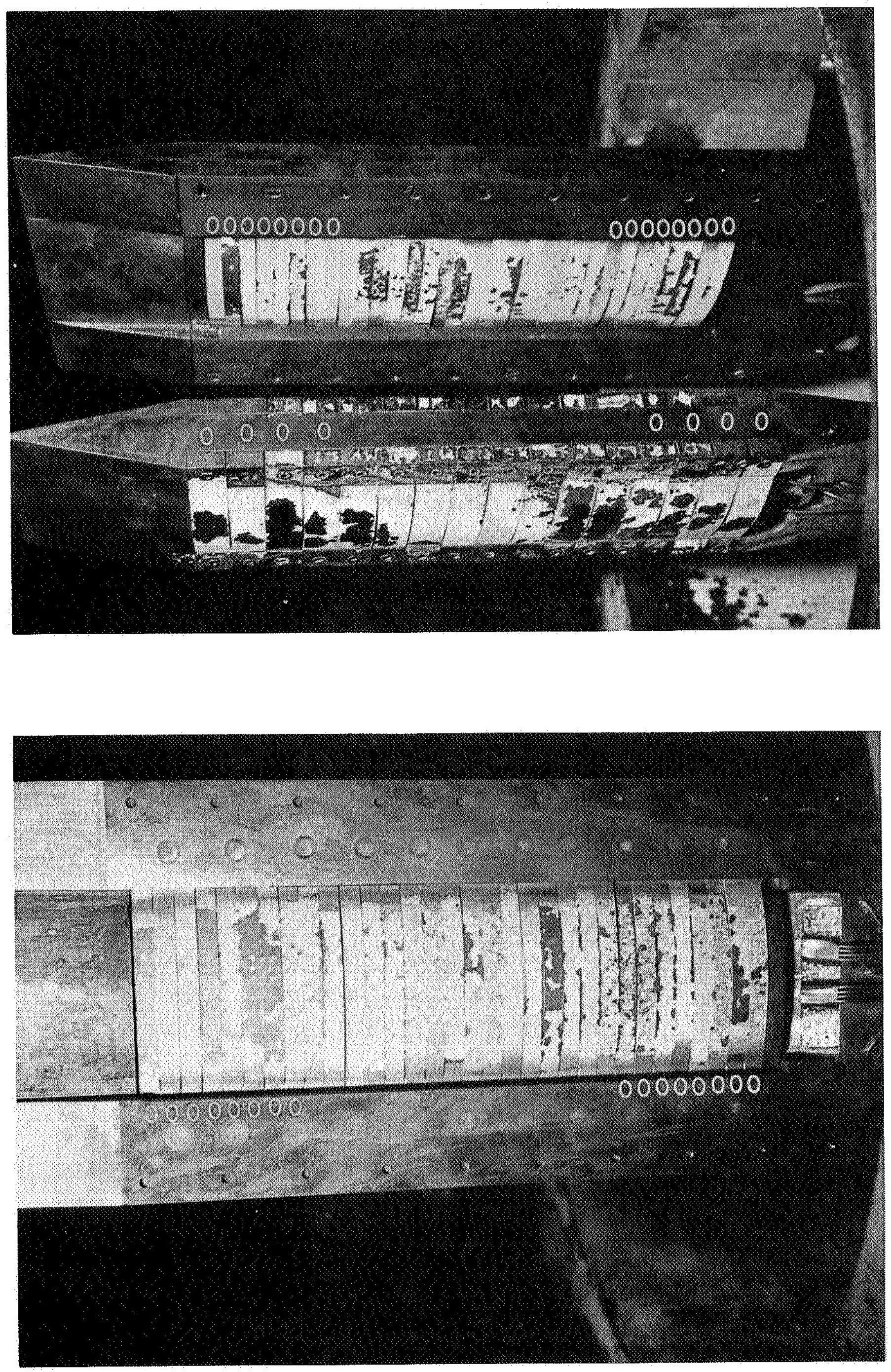


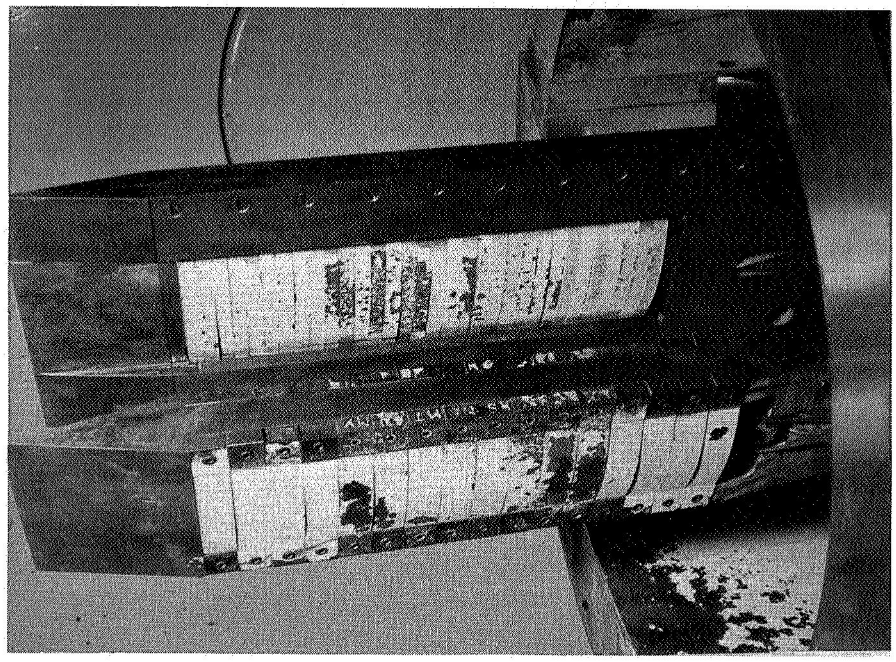
Figure 5-3 APPEARANCE OF TUNNEL SPECIMENS AFTER COMPLETION OF RUN II

NOTE: White oval denotes specimens removed and exchanged with new ones immediately after photograph was taken.

20 hrs duration Nominal specimen temp. 600°F

NASA inner surface

Lockheed outer surface



NASA outer surface

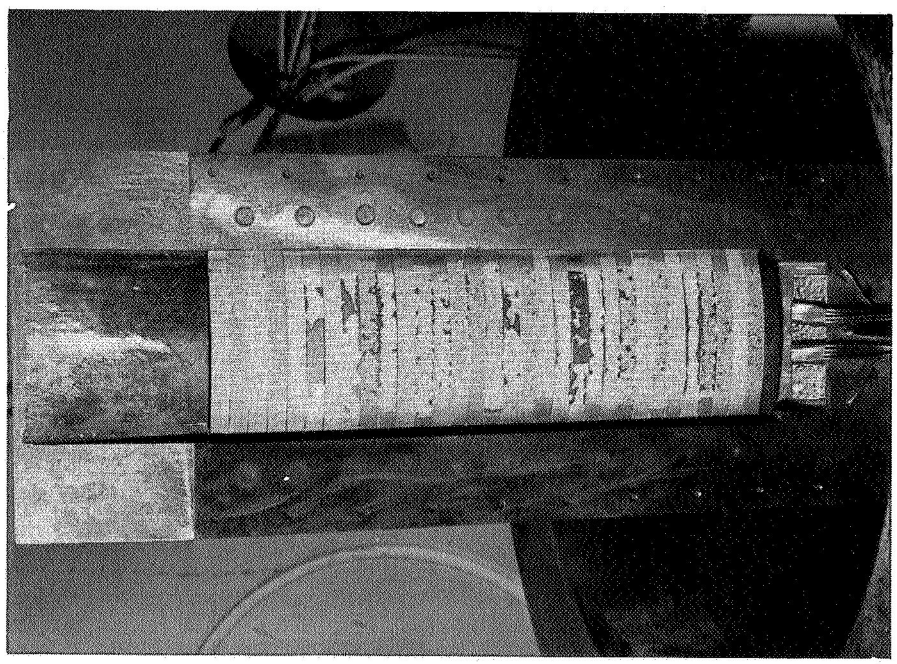


Figure 5-4 APPEARANCE OF TUNNEL SPECIMENS AFTER COMPLETION OF RUN III

salt loss occurred during the initial few minutes of each test. This suggests that the thermal shock and other transient phenomena associated with the tunnel start up may loosen the salt enough to allow its removal by the high velocity airstream.

In order to complete the documentation, immediately after being removed from the tunnel all specimens (with a single exception) were photographed to provide permanent records of the post test condition of the salt coatings. The photographs of the Lockheed and both surfaces of the NASA specimens are shown in Figs. 5-5 and 5-6. The particular position of the specimen on the plate is indicated. NASA positions are numbered from 1 through 32 beginning at the front (toward leading edge) of the mounting plate. Lockheed positions are numbered from 1 through 16 (front to back) and are designated as arising from the inner or outer surfaces of the plate. The individual specimen numbers and alloy compositions are also indicated.

From these photos it is clear that the specimens which underwent exposure at 700°F (15, 30, and 50 hours) experienced considerably more loss of salt than the final 20 hour 600°F specimens. It is particularly interesting to closely inspect areas of salt loss in the polished Lockheed specimens. It is evident that when the salt was lost from certain specimens, very shiny substrate metal was exposed. However, in other areas of salt loss, the exposed metal surface was rough and covered with corrosion products. One might suspect that perhaps in these shiny regions, the salt came off very early in the test before any appreciable corrosion had occurred. However, this assumption is not strictly correct as demonstrated by the fact that there were some specimens in which virtually no salt was removed during Run I but salt removal did occur during Run II, revealing uncorroded shiny metal. It is interesting to point out that virtually no stress corrosion cracking was observed in regions of salt loss when later examined under the microscope.

Inspection of the Lockheed photos (Figs. 5-5a and 5-5b) reveals that in general more salt loss occurred on the 30 and 50 hour than on the 15 hour specimens. For the NASA specimens however, (Figs. 5-6a and 5-6b), salt removal was somewhat more uniform occurring to a similar extent in all specimen groups.

No salt came off of the Lockheed oven test specimens; furthermore, these oven specimens displayed no such large uncorroded shiny regions. Thus, for the Lockheed tunnel specimens tested at 700°F, it is clear that the simulated supersonic flight conditions were instrumental in promoting removal of salt and very definitely reduced the extent of corrosion in certain regions of the surface. For the 600°F Lockheed tunnel specimens, however, very little salt removal occurred. In general, surface corrosion was very light for the 600°F tunnel tests, and the surfaces of these specimens remained quite uniformly shiny after exposure. Interestingly enough, a few of the NASA oven test specimens did experience some slight loss of salt. The reasons for the difference in salt adherence for the NASA and Lockheed oven test specimens were not readily apparent.

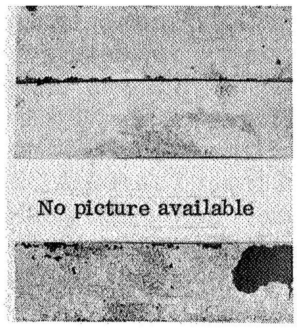
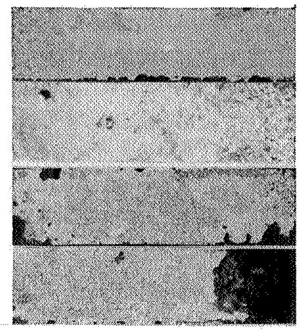
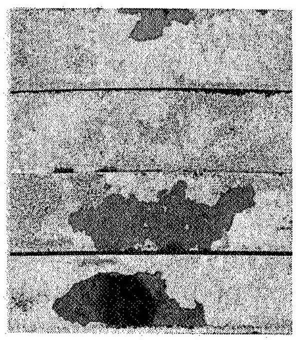
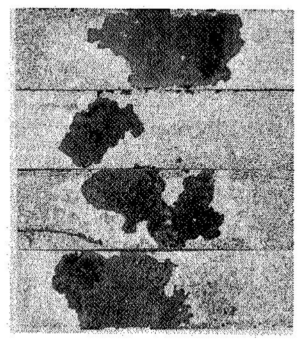
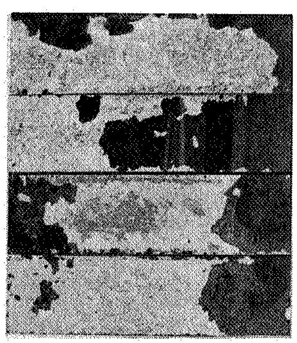
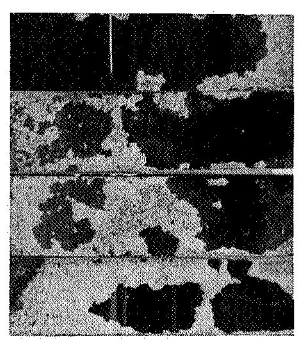
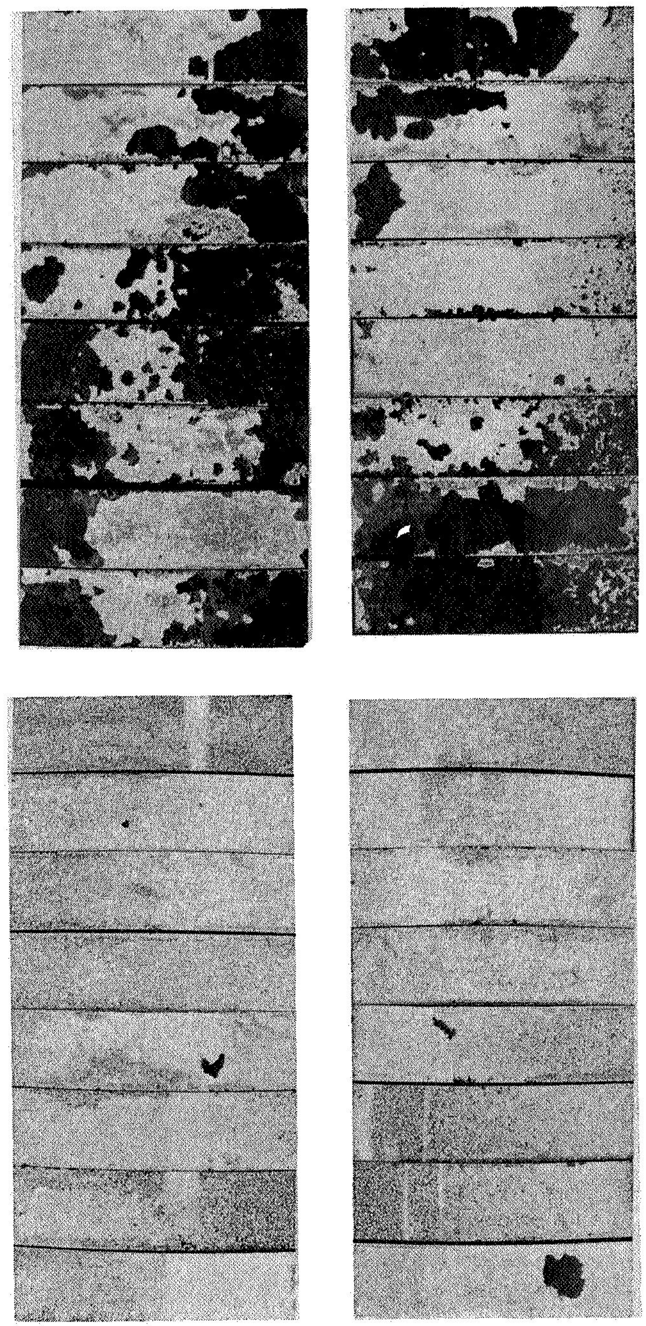
Test	Position on mounting plate	Inner surface	Outer surface	Alloy	Nominal temperature
First 15 hour	1			6-4	700°
	2			8-1-1	
	15	No picture available		6-4	
	16			8-1-1	
Second 15 hour	1			6-4	700°
	2			8-1-1	
	15			6-4	
	16			8-1-1	
30 hour	3			6-4	700°
	4			8-1-1	
	13			6-4	
	14			8-1-1	

Figure 5-5a APPEARANCE OF LOCKHEED TUNNEL SPECIMENS AFTER COMPLETION OF INDICATED TEST



Test	Position on mounting plate	Inner surface	Outer surface	Alloy	Nominal temperature
50 hour	5			6-4	30 hrs 700°F 20 hrs 600°F
	6			8-1-1	
	7			6-4	
	8			8-1-1	
	9			6-4	
	10			8-1-1	
	11			6-4	
	12			8-1-1	
20 hour	1			6-4	600°F
	2			8-1-1	
	3			6-4	
	4			8-1-1	
	13			6-4	
	14			8-1-1	
	15			6-4	
	16			8-1-1	

Figure 5-5b APPEARANCE OF LOCKHEED TUNNEL SPECIMENS AFTER COMPLETION OF INDICATED TEST

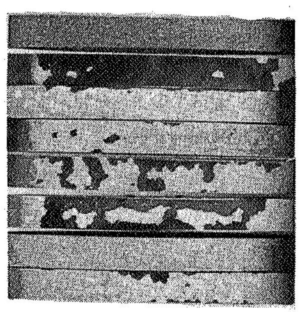
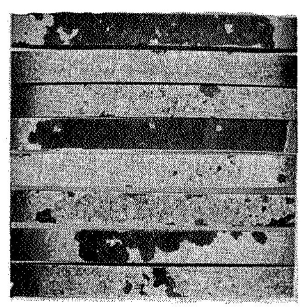
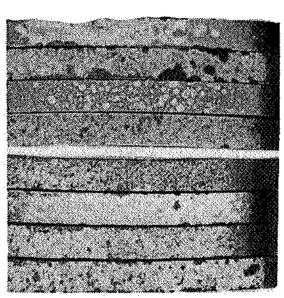
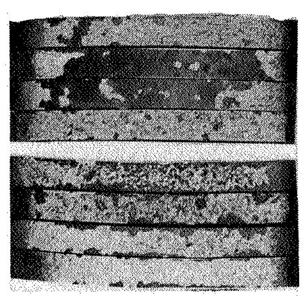
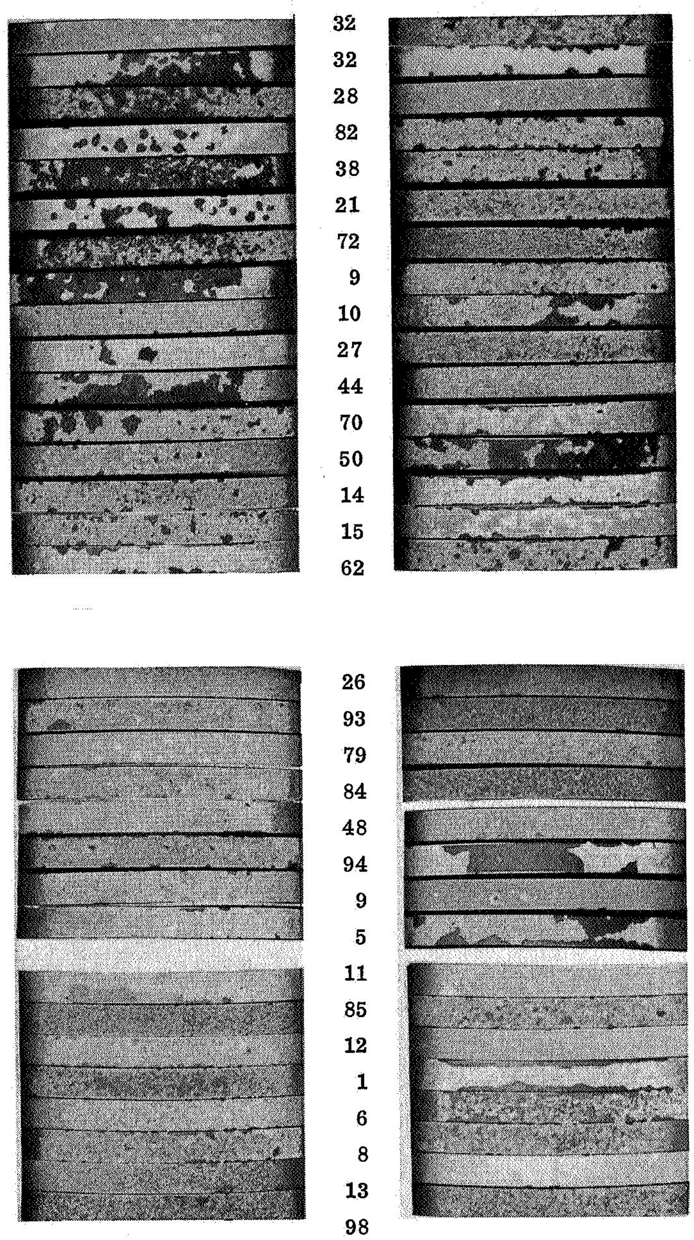
Test	Position on mounting plate	Inner surface	NASA specimens No.	Outer surface	Alloy	Nominal temperature
First 15 hour	1		25		6-4	700°F
	2		81		8-1-1	
	3		53		6-4	
	4		29		8-1-1	
	29		19		6-4	
	30		87		8-1-1	
	31		16		6-4	
	32		11		8-1-1	
Second 15 hour	1		29		6-4	700°F
	2		23		8-1-1	
	3		45		6-4	
	4		43		8-1-1	
	29		8		6-4	
	30		37		8-1-1	
	31		18		6-4	
	32		19		8-1-1	
30 hour	5		7		6-4	700°F
	6		46		8-1-1	
	7		54		6-4	
	8		55		8-1-1	
	25		43		6-4	
	26		97		8-1-1	
	27		27		6-4	
	28		28		8-1-1	

Figure 5-6a APPEARANCE OF NASA TUNNEL SPECIMENS AFTER COMPLETION OF INDICATED TEST



Test	Position on mounting plate	Inner surface	NASA specimen no.	Outer surface	Alloy	Nominal temperature
	9		32		6-4	
	10		32		8-1-1	
	11		28		6-4	
	12		82		8-1-1	
	13		38		6-4	
	14		21		8-1-1	
	15		72		6-4	
	16		9		8-1-1	30 hrs 700°F
50 hour	17		10		6-4	20 hrs 600°F
	18		27		8-1-1	
	19		44		6-4	
	20		70		8-1-1	
	21		50		6-4	
	22		14		8-1-1	
	23		15		6-4	
	24		62		8-1-1	
	1		26		6-4	
	2		93		8-1-1	
	3		79		6-4	
	4		84		8-1-1	
	5		48		6-4	
	6		94		8-1-1	
	7		9		6-4	
20 hour	8		5		8-1-1	600°F
	25		11		6-4	
	26		85		8-1-1	
	27		12		6-4	
	28		1		8-1-1	
	29		6		6-4	
	30		8		8-1-1	
	31		13		6-4	
	32		98		8-1-1	

Figure 5-6b APPEARANCE OF NASA TUNNEL SPECIMENS AFTER COMPLETION OF INDICATED TEST

Analysis of Lockheed Specimens

After being tested, all Lockheed specimens were cleaned, etched and examined for cracking damage under 400X magnification. Crack severity data were obtained for all specimens using the counting techniques described earlier (refer to Section 2). On each specimen, crack intersection counts were made at the following locations: in the center, and at 5 mm and 10 mm on either side of center. In addition, the positions of the last cracks toward the ends were determined to establish the stress threshold point. For each specimen, a calculation was performed to determine the stress distribution curve and the stress relaxation which occurred during the test.

Initial and relaxed stresses. - A significant amount of permanent set from stress relaxation was observed. The stress relief which occurred was largely the result of the combination of the high stress levels and high temperatures used in these tests. Table 5-1 presents the results of the analysis made to determine the average initial stresses and the final relaxed stress for the Lockheed specimens.

TABLE 5-1

INITIAL AND FINAL STRESSES
LOCKHEED TUNNEL SPECIMENS

Material	Nominal temp °F	Exposure hrs	Av. max. estimated stress KSI		
			Initial	Relaxation	Final
Ti-8Al-1Mo-1V	700	15	78.0	7.3	70.7
		30	77.6	9.5	68.1
	700/600	30/20	81.3	10.9	70.4
	600	20	82.5	2.2	80.3
Ti-6Al-4V	700	15	73.9	9.2	64.7
		30	76.6	10.3	63.3
	700/600	30/20	80.7	12.1	68.6
	600	20	83.7	1.2	82.5

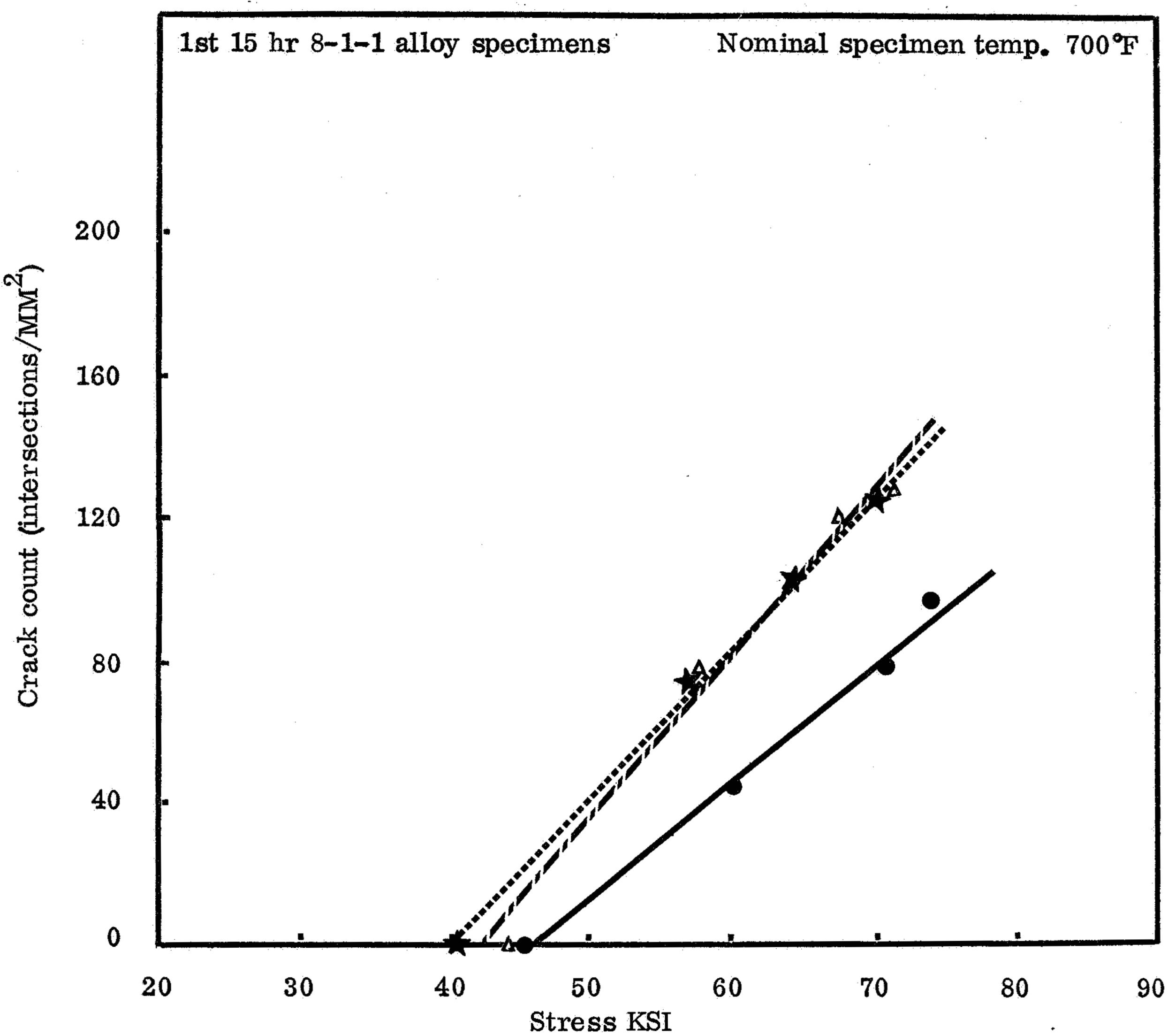
As would be expected longer exposures and higher temperatures resulted in higher values of stress relaxation. Also of interest is that the second 15 hour exposure (on the 30 hour test specimens) resulted in a much smaller increment of stress relief than did the first 15 hours.

In order to determine the possible effect of salt cracking on the stress relaxation measurements, a comparison was made of the permanent deflection for uncoated oven specimens with those which had been coated. The data all fell within the same scatter band and no significant difference could be determined. This indicates that for the test conditions utilized in this program, the presence of salt cracks plays an insignificant roll in determining permanent set of the Lockheed specimens.

Results of the crack count. - After the stress distribution for each specimen was determined, the crack intersection count data were then correlated with the stress level at each point of interest. When these crack count data were plotted versus stress level, considerable specimen-to-specimen data scatter was observed. This was even true for specimens which had been exposed under essentially identical oven test conditions. This result is not unexpected since fairly large amounts of data scatter are commonly observed in hot salt corrosion studies. The scatter problem may be alleviated considerably by averaging corresponding data obtained from specimens tested under similar exposure conditions. When this averaging process was applied to these data, reasonably good correlations between cracking severity and stress level were obtained. Average data plots of this type were prepared for each set of test conditions and are presented in Figs. 5-7 through 5-16*. The number of specimens involved in each test is indicated; in those cases where very few specimens were involved, some caution should be exercised in interpreting the data, since the number used may have been insufficient to completely average out normal scatter.

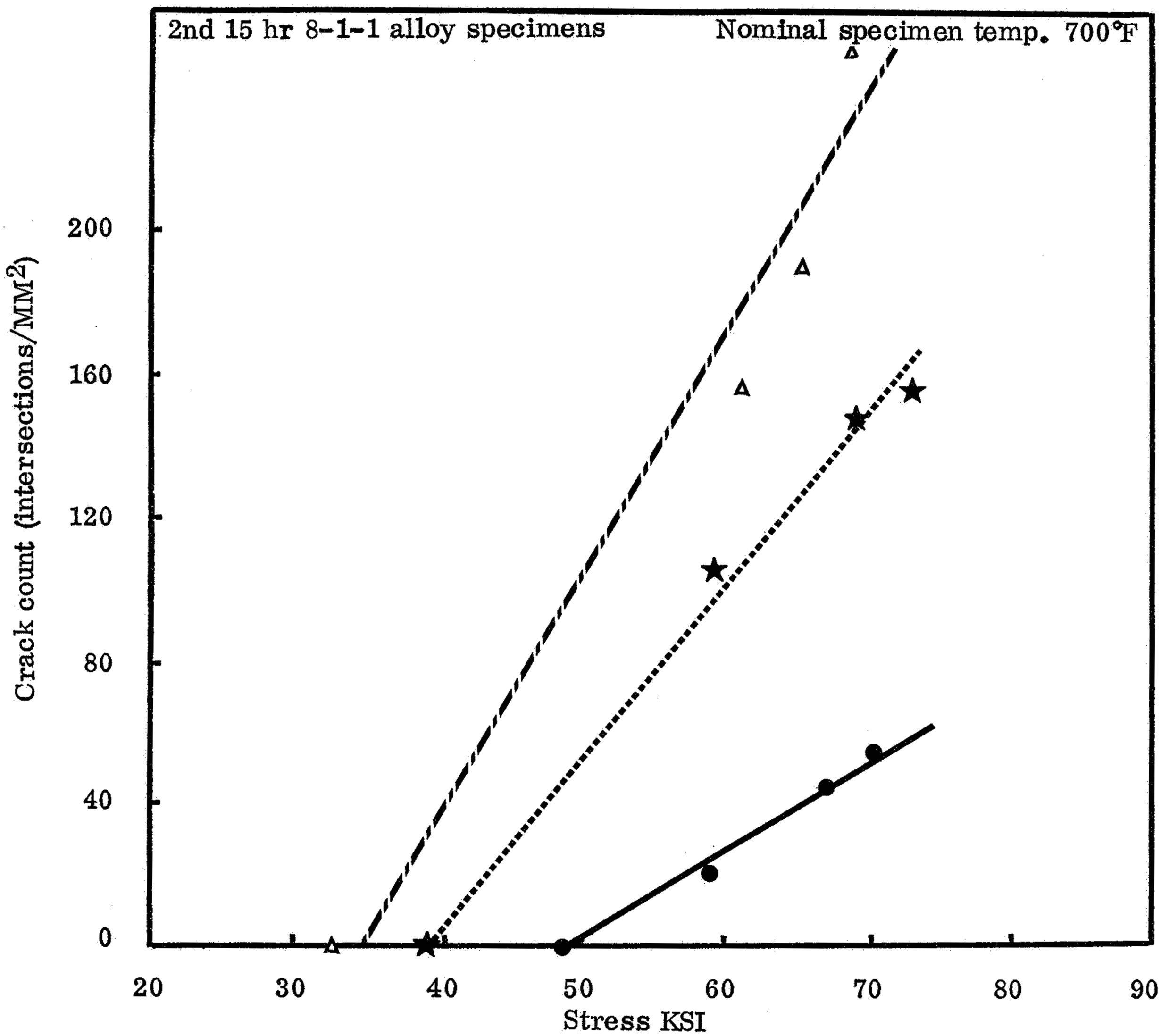
Stress threshold. - It should be recalled that cracking data plots provide two distinctly different types of information concerning the titanium stress corrosion process. The individual slopes of the lines provide a means of comparing relative cracking damage between individual sets of specimens, and the horizontal axis intercept point is a measure of the stress threshold value for the particular test specimens involved. For convenience, all of the intercept values obtained from these Figures are presented in Table 5-2. An examination of these data indicate that for all specimens (except the 600°F tunnel test on the 6-4 alloy) the threshold values all fall within the scatter band from 34 KSI to 48 KSI. Hence, there is no obvious correlation between these values and temperature, time, type of exposure, or type of alloy. (It is noted, however, that for the 8-1-1 alloy the tunnel test did give higher values than the oven test but no such correlation was apparent for the 6-4 alloy.) The damage suffered by the 6-4 alloy specimens exposed at 600°F in the tunnel was so small that no threshold value could be established.

*Data from all test specimens except the uncoated oven test specimens are included in these Figures. No cracking whatever was observed on the uncoated test specimens.



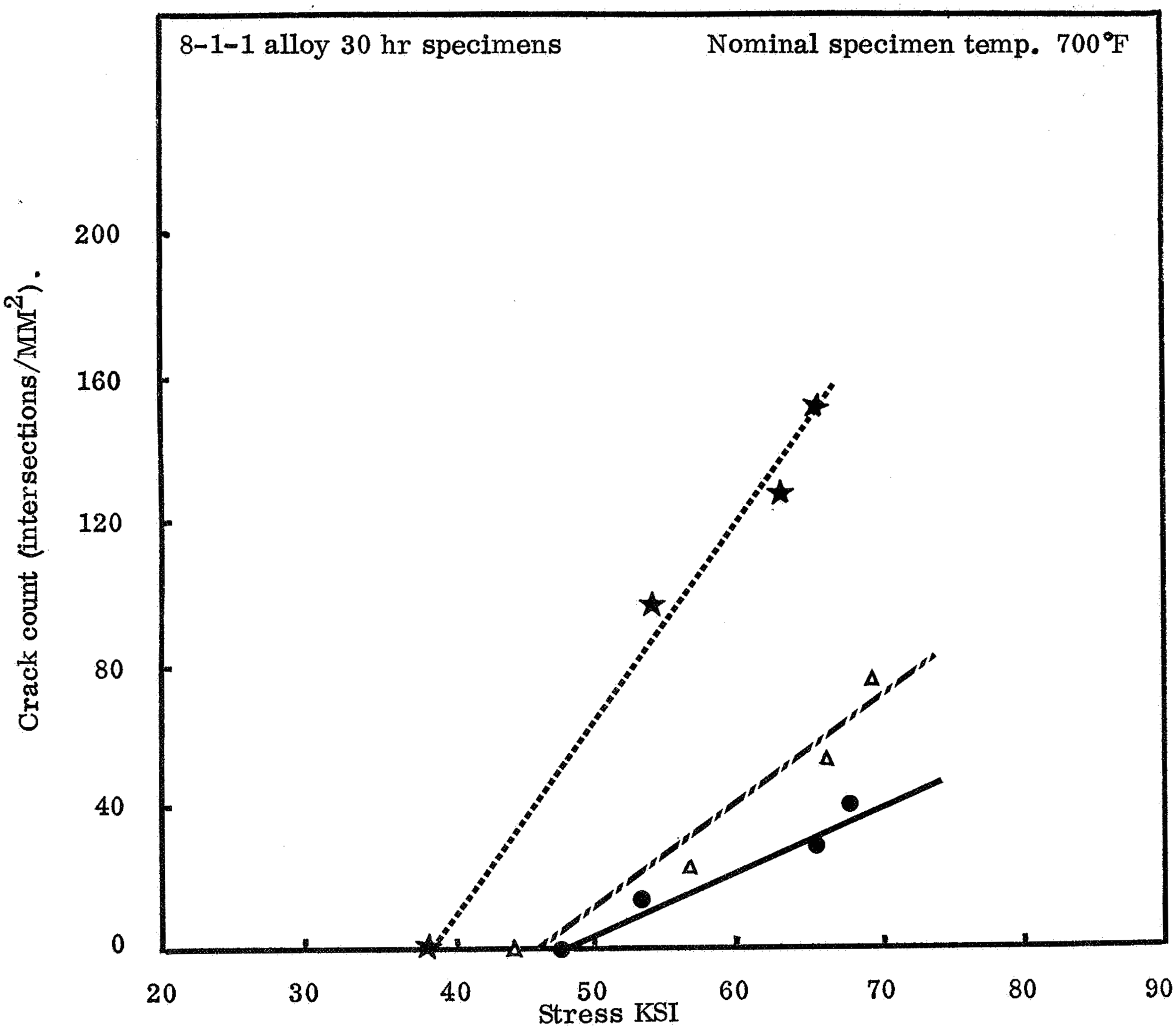
	Test	Number of specimens	Threshold (KSI)	Line slope
——●——	Tunnel	4	46	3.2
······★······	Corres. oven	4	40	4.1
— — △ — —	Avg. temp. oven	2	43	4.6

Figure 5-7 CRACK INTERSECTION COUNT AS A FUNCTION OF STRESS LEVEL



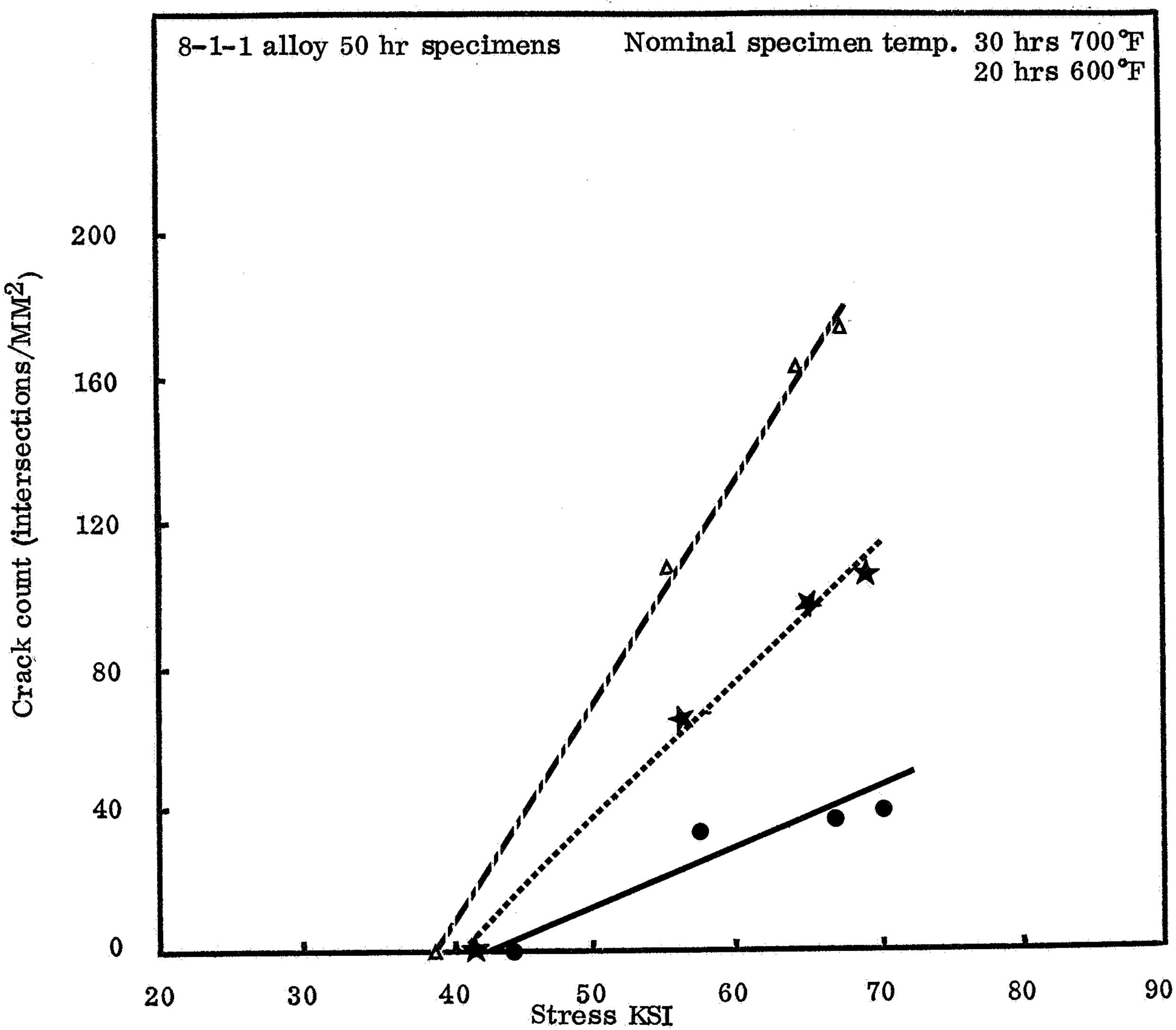
Test	Number of specimens	Threshold (KSI)	Line slope
Tunnel	4	48	2.5
Corres. oven	4	39	4.8
Avg. temp. oven	2	34	6.6

Figure 5-8 CRACK INTERSECTION COUNT AS A FUNCTION OF STRESS LEVEL



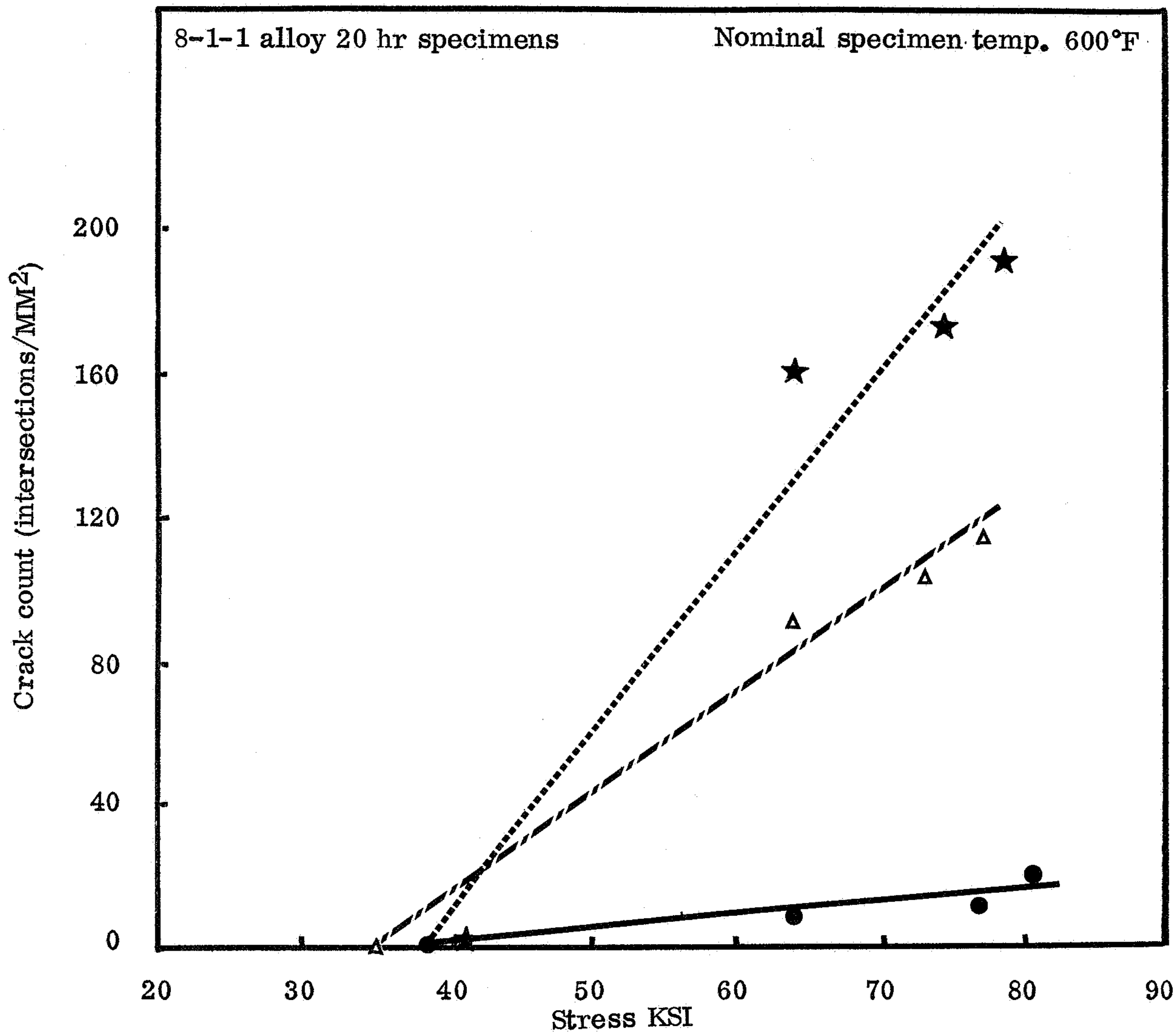
	Test	Number of specimens	Threshold (KSI)	Line slope
——●——	Tunnel	4	48	1.7
·····★·····	Corres. oven	4	39	5.5
–·–△–·–	Avg. temp. oven	2	46	2.9

Figure 5-9 CRACK INTERSECTION COUNT AS A FUNCTION OF STRESS LEVEL



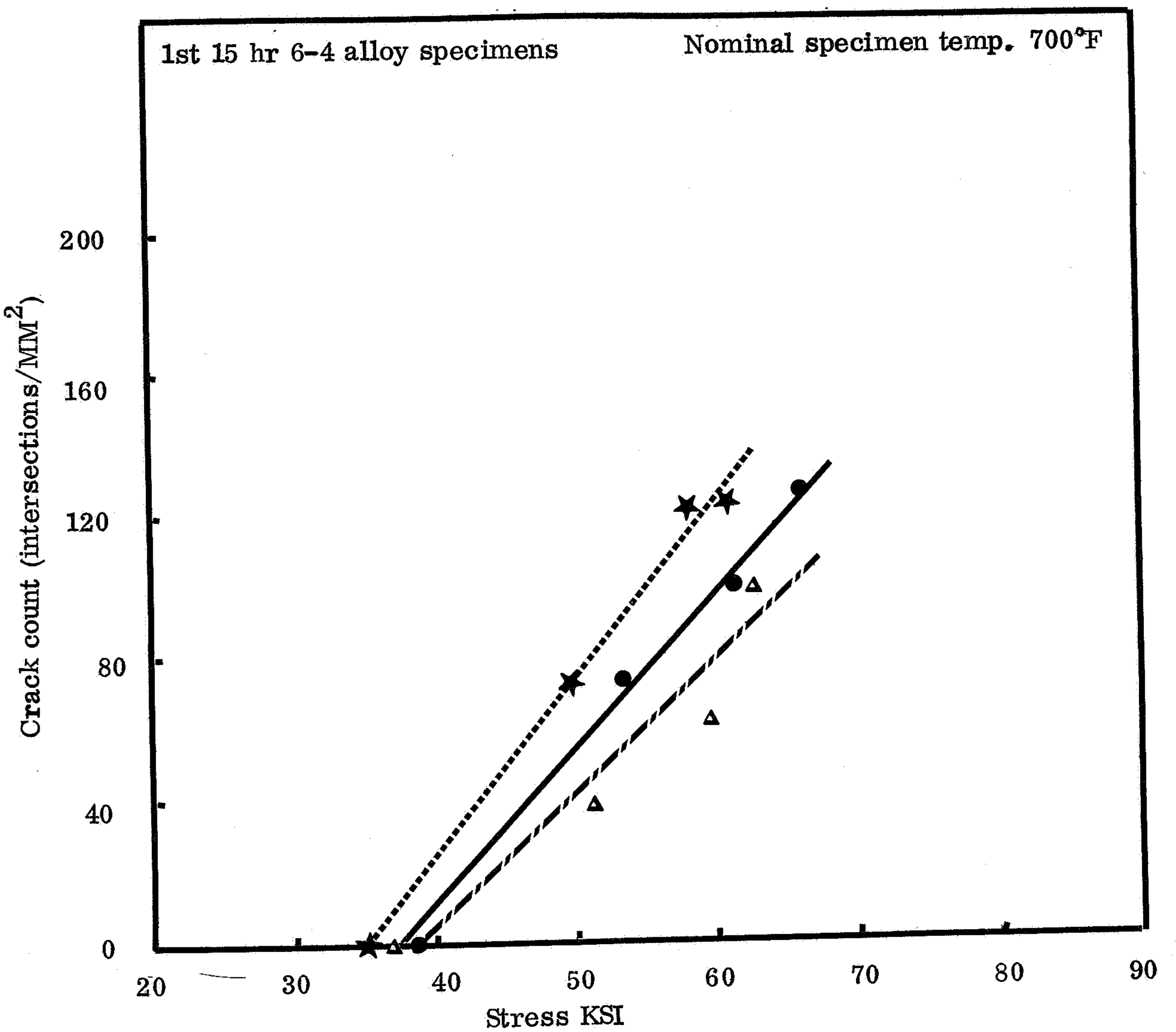
	Test	Number of specimens	Threshold (KSI)	Line slope
	Tunnel	8	43	1.7
	Corres. oven	8	41	3.8
	Avg. temp. oven	6	39	6.1

Figure 5-10 CRACK INTERSECTION COUNT AS A FUNCTION OF STRESS LEVEL



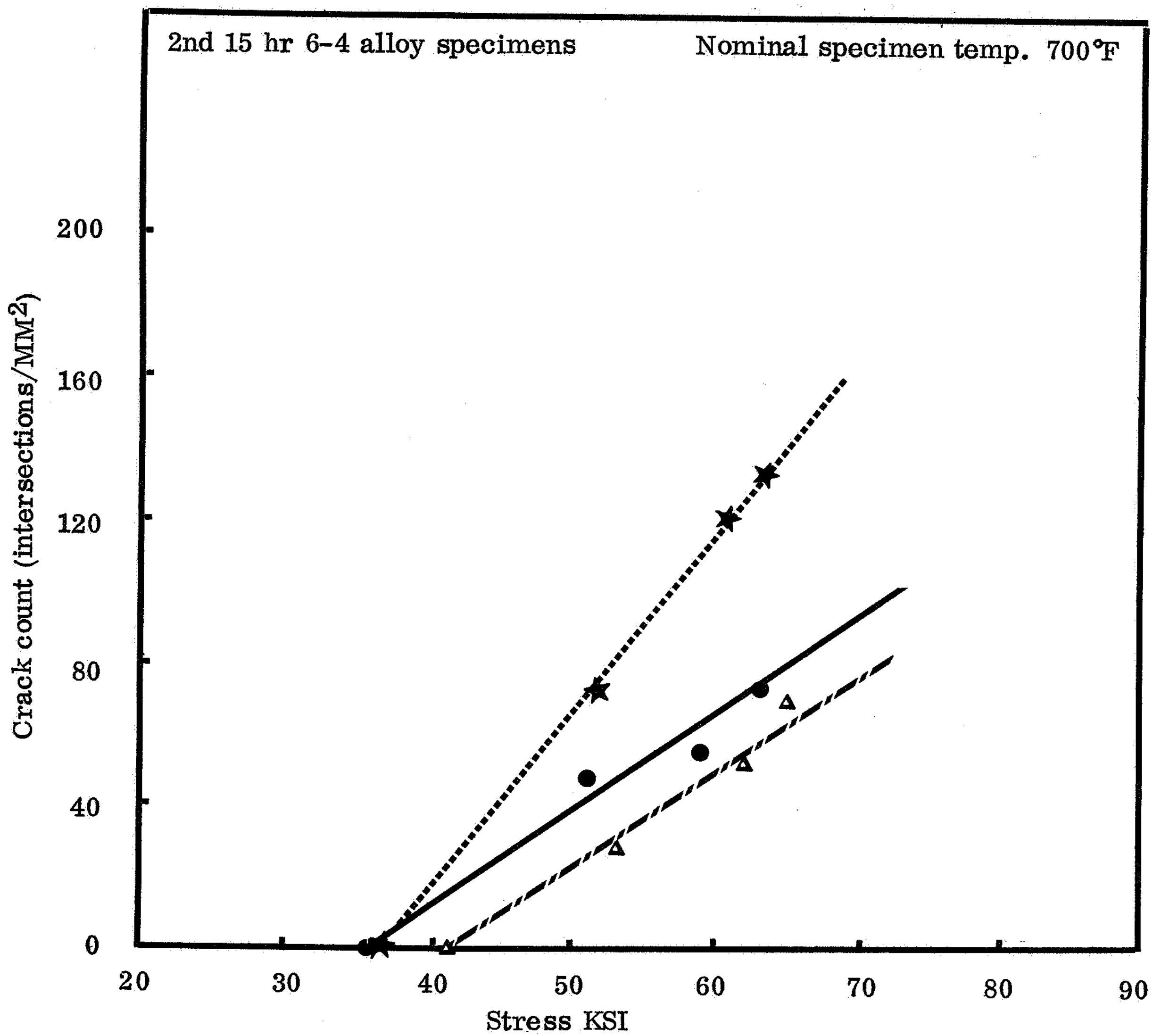
	Test	Number of specimens	Threshold (KSI)	Line slope
——●——	Tunnel	8	38	0.4
······★······	Corres. oven	8	38	5.1
–·–△–·–	Avg. temp. oven	4	35	2.7

Figure 5-11 CRACK INTERSECTION COUNT AS A FUNCTION OF STRESS LEVEL



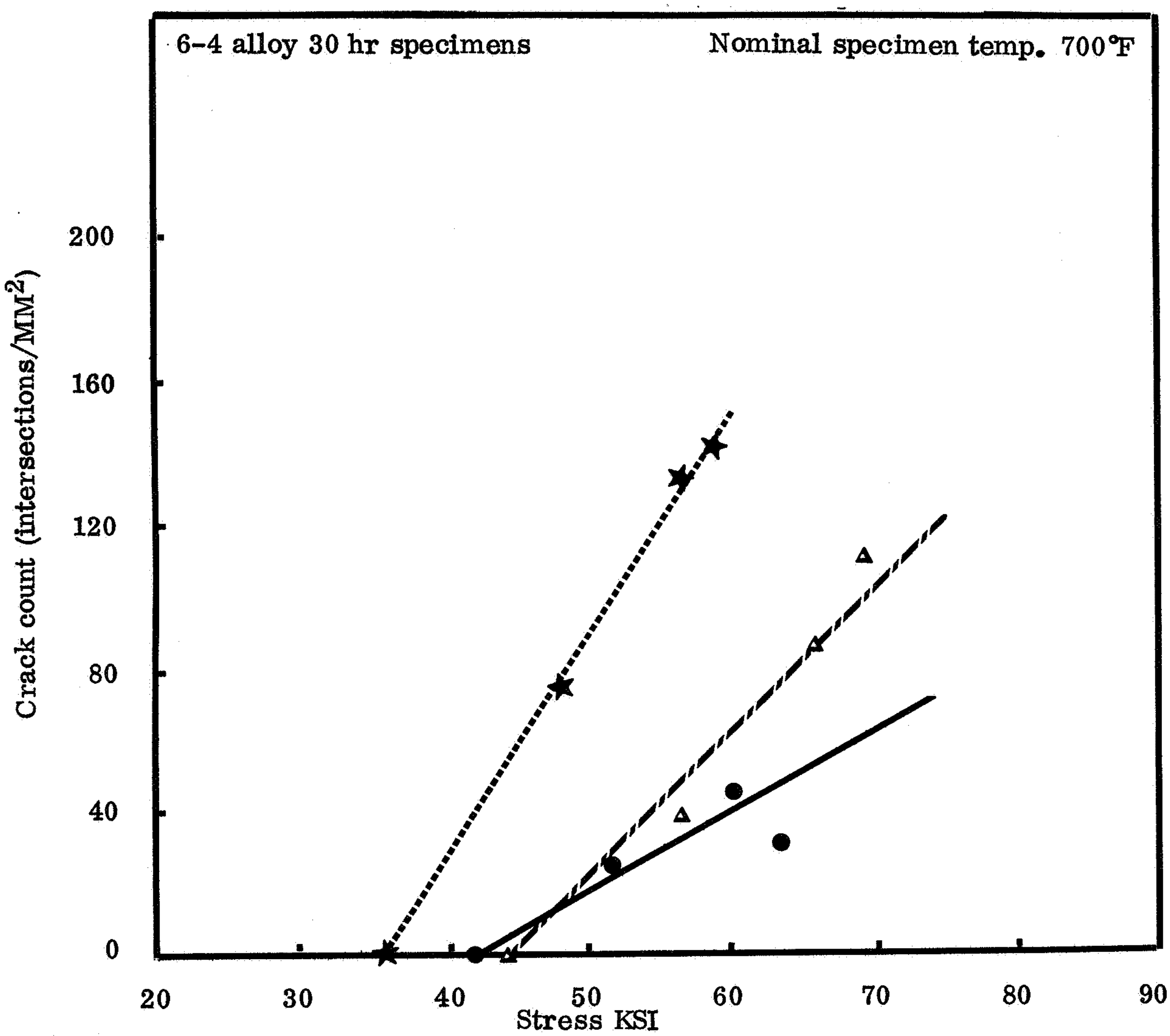
Test	Number of specimens	Threshold (KSI)	Line slope
Tunnel	4	38	4.4
Corres. oven	4	35	5.0
Avg. temp. oven	2	39	3.7

Figure 5-12 CRACK INTERSECTION COUNT AS A FUNCTION OF STRESS LEVEL



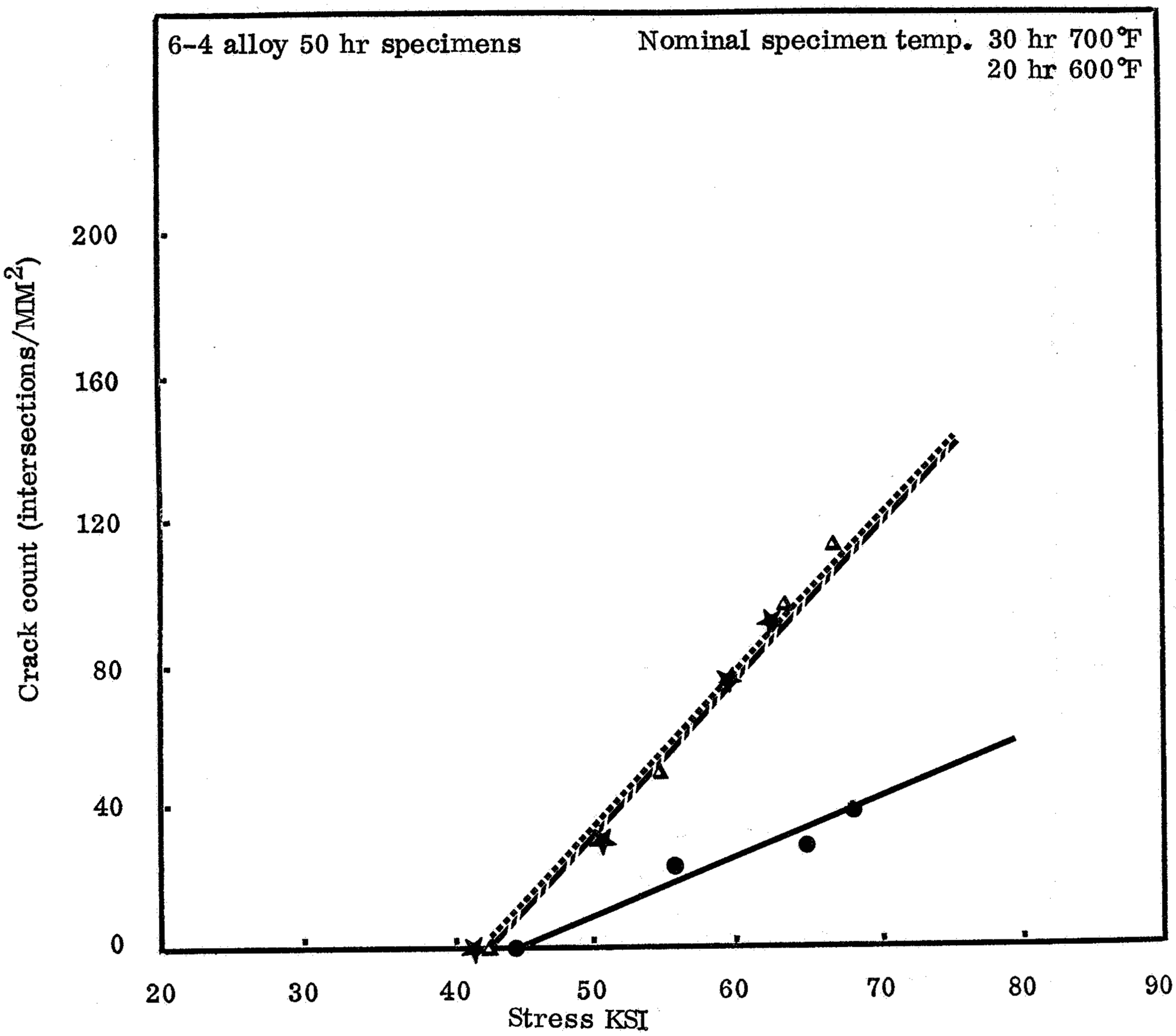
	Test	Number of specimens	Threshold (KSI)	Line slope
——●——	Tunnel	4	36	2.7
·····★·····	Corres. oven	4	37	4.9
–·–△–·–	Avg. temp. oven	2	41	2.5

Figure 5-13 CRACK INTERSECTION COUNT AS A FUNCTION OF STRESS LEVEL



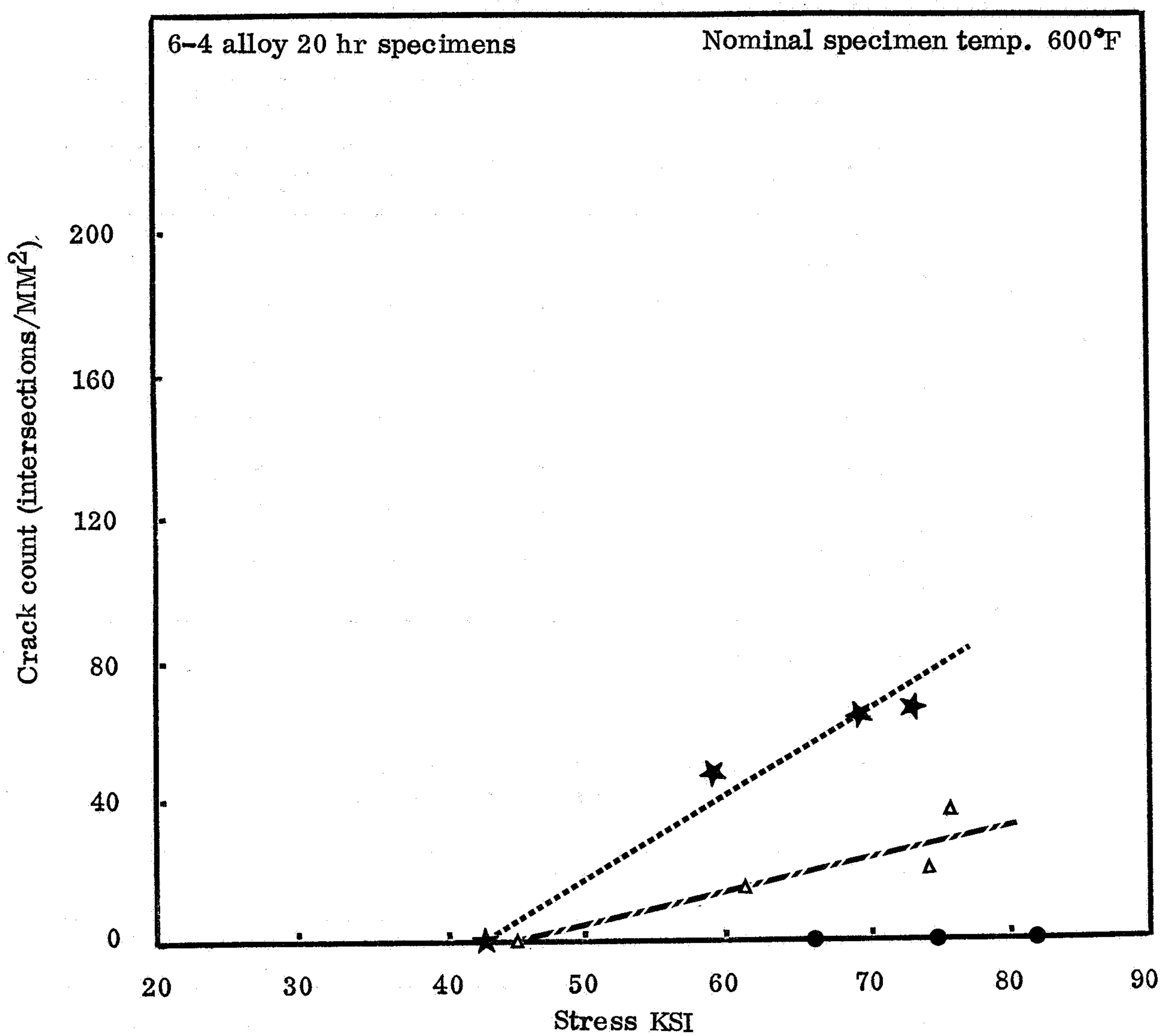
	Test	Number of specimens	Threshold (KSI)	Line slope
	Tunnel	4	42	2.2
	Corres. oven	4	36	6.1
	Avg. temp. oven	2	44	3.9

Figure 5-14 CRACK INTERSECTION COUNT AS A FUNCTION OF STRESS LEVEL



	Test	Number of specimens	Threshold (KSI)	Line slope
——●——	Tunnel	8	45	1.6
······★······	Corres. oven	8	43	4.3
–·–·△·–·–	Avg. temp. oven	6	43	4.3

Figure 5-15 CRACK INTERSECTION COUNT AS A FUNCTION OF STRESS LEVEL



	Test	Number of specimens	Threshold (KSI)	Line slope
——●——	Tunnel	8	--	0.0
······★······	Corres. oven	8	43	2.3
–·–△–·–	Avg. temp. oven	4	45	0.9

Figure 5-16 CRACK INTERSECTION COUNT AS A FUNCTION OF STRESS LEVEL

TABLE 5-2

EXPERIMENTALLY DETERMINED STRESS THRESHOLD VALUES
FOR EACH GROUP OF LOCKHEED TEST SPECIMENS

	Specimen group	1st 15 hr	2nd 15 hr	30 hr	50 hr	20 hr
	Nominal temp. °F	700	700	700	30 hrs 700 20 hrs 600	600
8-1-1 alloy stress threshold values (KSI)	Tunnel tests	46	48	48	43	38
	Corresponding oven test	40	39	39	41	38
	Constant temp. oven test	43	34	46	39	35
6-4 alloy stress threshold values (KSI)	Tunnel tests	38	36	42	45	*
	Corresponding oven test	35	37	36	43	43
	Constant temp. oven test	39	41	44	43	45

*Insufficient cracking to establish a threshold

TABLE 5-3

SLOPE OF CRACK INTERSECTION COUNT LINES FOR EACH
CORRESPONDING GROUP OF LOCKHEED TEST SPECIMENS

	Specimen group	1st 15 hr	2nd 15 hr	30 hr	50 hr	20 hr
	Nominal temp. °F	700	700	700	30 hrs 700 20 hrs 600	600
Line slopes for 8-1-1 alloy specimens counts/(MM^2) KSI	Tunnel tests	3.2	2.5	1.7	1.7	0.4
	Corresponding oven test	4.1	4.8	5.5	3.8	5.1
	Constant avg. temp. oven test	4.6	6.6	2.9	6.1	2.7
Line slopes for 6-4 alloy specimens counts/(MM^2) KSI	Tunnel tests	4.4	2.7	2.2	1.6	0.0*
	Corresponding oven test	5.0	4.9	6.1	4.3	2.3
	Constant avg. temp. oven test	3.7	2.5	3.9	4.3	0.9

*Insufficient cracking to produce significant line slope value

Corrosion cracking damage. - A very significant feature of Figs. 5-7 through 5-16 is revealed by the slopes of the individual lines. A severely cracked specimen will yield a line with a steeper slope than one with less extensive cracking. The slopes of all these lines have been determined and these data are presented in Table 5-3. From this table it is observed that for both alloys the tunnel tests in general produced less cracking damage than the oven tests. This difference is particularly striking for the 20 hr 600°F tests. It is also noted that there is no significant difference between the damage experienced by the two alloys when exposed at 700°F, but that 600°F exposure was less damaging on the 6-4 than on the 8-1-1 alloy.

It is of interest to note an apparent anomaly in the data in that the tunnel test specimens exposed for longer periods of time at 700°F had fewer cracks. This inverse correlation is attributed to the effects of salt removal; these effects will be discussed in detail in the next section. The oven tests showed no consistent correlation of cracking with time of exposure, indicating that all or most of the damage occurred within the first 15 hrs of exposure. This would also help explain the lack of time dependence for the tunnel data.

The cracking data for the Lockheed 600°F tests is very significant in that the 600°F oven tests produced cracking which was considerably more severe than the corresponding tunnel tests. It is obvious that there were differences in the test conditions between the tunnel and oven which were responsible for this effect. These differences could have been related to air flow, pressure, vibration, humidity, salt application techniques, etc, but this program did not provide specific information which would explain this result.

Effect of salt removal. - It has been pointed out that Lockheed 700°F specimens with longer exposure times had significantly fewer cracks. It would normally be expected that the opposite result would occur. However, microscopic examination of these specimens provided a clue to the reason for this apparently anomalous result. The examination revealed that regions of specimens which underwent salt removal had essentially no cracking damage. It is obvious that the absence of cracks in these regions will lower the overall specimen crack count values and result in lower line slopes. The 30 and 50 hour specimens, which were near the center of the mounting plate, underwent larger amounts of salt removal, and these specimens also yielded lower crack counts. From this it can only be concluded that the reduced cracking damage in these specimens was a result of salt loss.

In order to substantiate this correlation of cracking damage with salt removal, the percentage of the surface still covered with salt after testing was estimated by placing a transparent grid work of small squares over the photographs in Fig. 5-5 and counting squares on each specimen to determine the relative areas covered and uncovered with salt. These salt retention values were averaged for each of the 15, 30, and 50 hour groups and plotted versus the corresponding line slopes as shown in Fig. 5-17. Since no salt was removed from the Lockheed specimens during the oven tests, the slopes of the 15, 30 and 50 hour hour oven test lines were averaged and the resulting mean value was

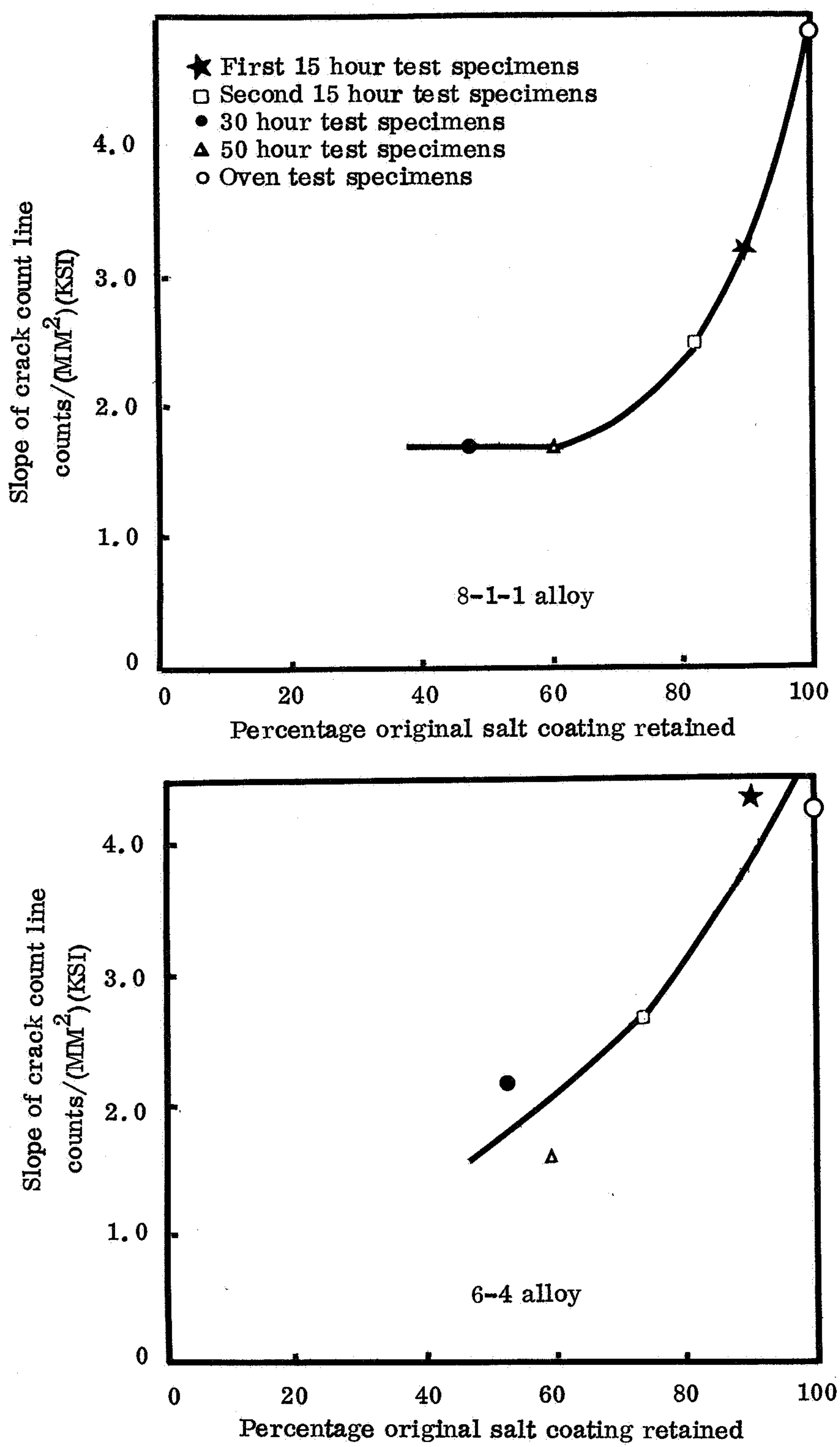


Figure 5-17 AVERAGE SLOPE OF CRACK COUNT LINES AS A FUNCTION OF THE AMOUNT OF SALT RETAINED AT COMPLETION OF TEST

plotted versus the 100% salt retention point. Notice that for the 8-1-1 alloy an excellent correlation between line slope and salt retention is indicated. For the 6-4 alloy the curve fit is not as good, but a similar type correlation is suggested.

Analysis of NASA Self-Stressed Specimens

Initial and relaxed stresses. - In order to ensure stress corrosion cracking during the short tunnel exposures, the NASA specimens were designed for outer fiber bending stresses of about 80 to 90 KSI at 600°F to 700°F. The dimensions of the specimens were kept constant so as to obtain a reasonably smooth aerodynamic surface when mounted in the tunnel. Consequently, because of the difference in moduli between the 6-4 and the 8-1-1 titanium, the maximum fiber stress in specimens of the former material is less than for the latter. The stress relaxation was estimated from the residual curvature of strips cut from specimens exposed in the tunnel and ovens at 600°F and 700°F. The chord length and mid-ordinate height of these strips were measured, and the equivalent elastic stresses required to produce this curvature were calculated. The estimated initial, relaxation, and final stresses are summarized in Table 5-4.

TABLE 5-4

INITIAL AND FINAL STRESSES
NASA SPECIMENS

Material	Nominal temp. °F	Exposure hrs	Av. max. estimated stress KSI		
			Initial	Relaxation	Final
	700	15	86.5	23.2	63.3
Ti-8Al-1Mo-1V		30	86.5	26.4	60.1
	600	20	89.8	10.1	79.7
	700	15	75.5	21.1	54.4
Ti-6Al-4V		30	75.5	23.1	52.4
	600	20	80.3	4.7	75.6

Note that the final stresses at 600°F are 79.7 and 75.6 KSI for 8-1-1 and 6-4 alloys respectively, and that after 30 hours at 700°F these corresponding stresses are 60.1 and 52.4 KSI. As expected, higher temperatures and longer exposure times produced more stress relaxation; also the second 15 hours of exposure (for the

.30 hour specimens) produced a smaller increment of relaxation than did the first 15 hours.

Stress corrosion damage. - The NASA compressive method of testing specimens provides data which are indicative of the effect the stress corrosion process has on the loss of bending ductility and mechanical strength of the specimens. This reduction in properties is determined by the amount of shortening experienced by the compressed specimen at the time of failure and relating this shortening to that which occurs in uncorroded specimens. Hence the ratio of these two values, called relative shortening, provides a measure of the damage resulting from the test exposure. The higher the value of the relative shortening the less has been the damage experienced by the specimen.

Figure 5-18 summarizes the relative shortening data obtained on the NASA specimens. From this figure, taking into account the data scatter effects and relative number of specimens in each test group, the following observations are made.

1. The 8-1-1 alloy is more sensitive to damage in these environments than is the 6-4 alloy as evidenced by the fact that the relative shortening values for the 8-1-1 are significantly less than those for the corresponding 6-4 specimens. This result is in relative agreement with past experience which has shown that for the same test conditions the 8-1-1 alloy is more susceptable to stress corrosion damage and forms deeper cracks than does the 6-4 alloy (Ref. 4). However, it should be pointed out that the magnitude of this difference may have been influenced by the higher stress levels in the 8-1-1 specimens.

2. At 700°F it appears that the damage which occurred in the first 15 hour exposure period controls the results of the tests, i.e., any additional cracking which may occur during subsequent exposures does not significantly affect the test results.

3. For tests at 700°F there were no consistent differences in cracking damage between the tunnel and the oven test specimens. However, at the 600°F temperature, the tunnel specimens experienced considerably less damage than the oven test specimens.

4. Oven exposure at 600°F produced damage essentially as great as the 700°F exposure.

5. The extent of cracking damage which occurred in the two different types of oven tests was similar.

6. In general, data from the average temperature oven tests tended to produce less scatter than data from the directly corresponding oven tests. This may be due to the fact that specimens in the directly corresponding oven tests were run in groups at several slightly different temperatures, whereas the average temperature oven specimens were exposed at only one temperature value.

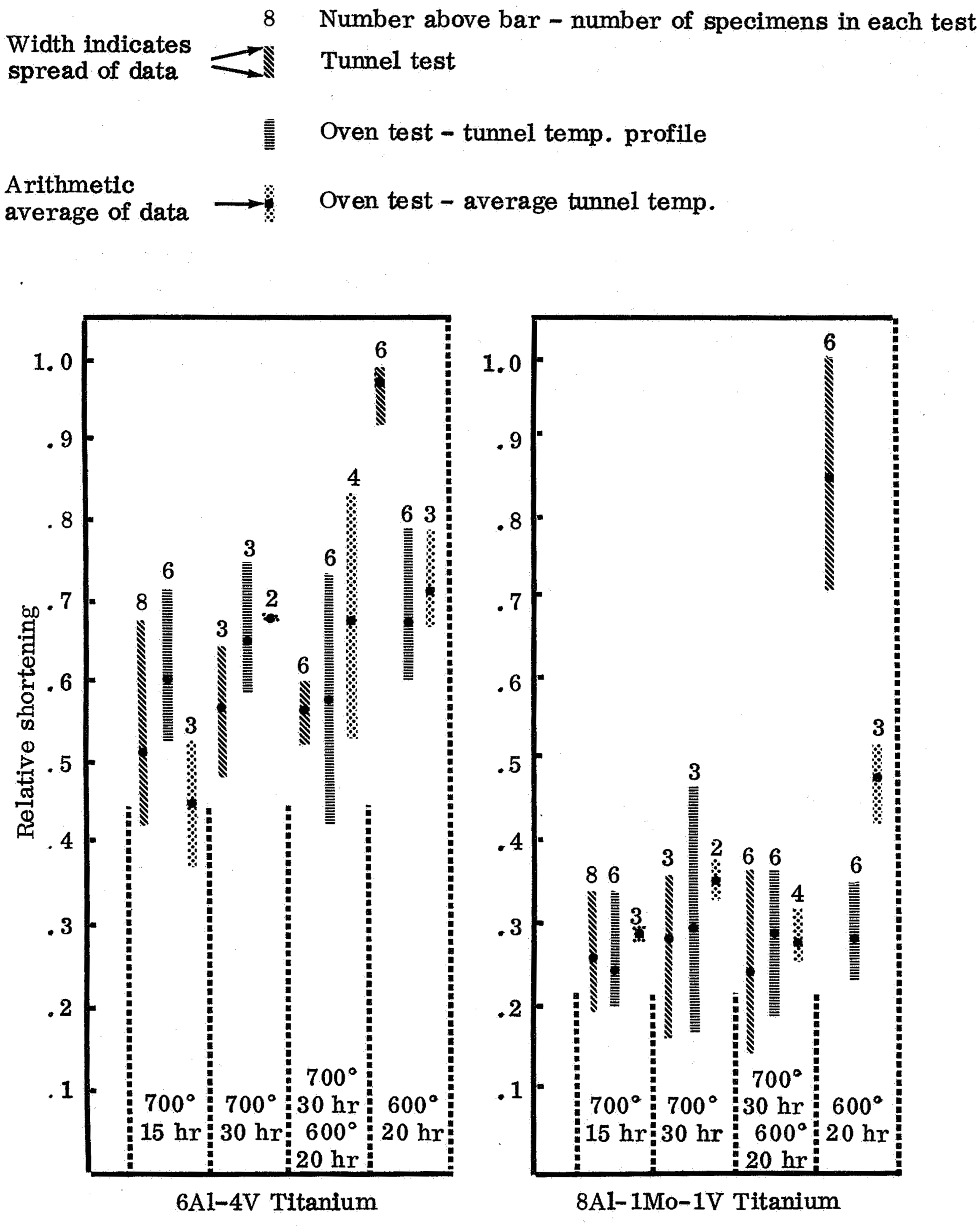


Figure 5-18 SHORTENING OF SALTED NASA SPECIMENS RELATIVE TO SHORTENING OF UNSALTED SPECIMENS

Crack observations. - Figs. 5-19 and 5-20 show photomicrographs of the surfaces of some of the NASA specimens. Fig. 5-19 can be used to compare the appearance of the 6-4 and 8-1-1 alloys when exposed in the tunnel and in the average temperature oven at 700°F. These figures do not reveal any specific reason for the differences between the results of the two alloys since the appearance and number of cracks appear comparable for both.

On the other hand, for the 600°F tests, Fig. 5-20 shows differences which in general correlate with the results of the tests shown in Fig. 5-18. Notice that in both the 600°F tunnel and oven tests the 6-4 alloy shows much less cracking than the 8-1-1 alloy. Also the 600°F oven test specimens exhibit more cracking than do the tunnel test specimens; this may account for the relative reduced strength for the oven tests, as indicated in Fig. 5-18. Care must be exercised not to extend these conclusions too far since they are based on a limited number of observations. More data would be required to completely verify these conclusions.

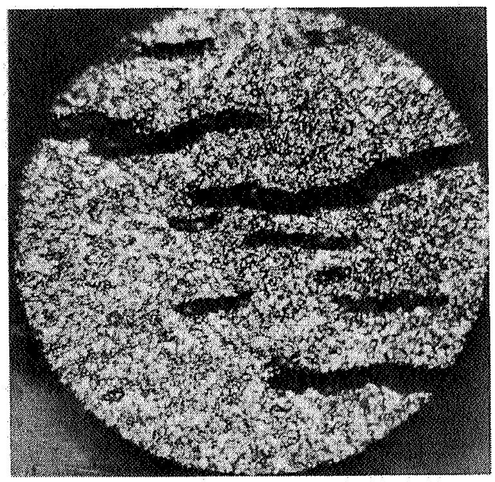
Section 6

CONCLUSIONS AND RECOMMENDATIONS

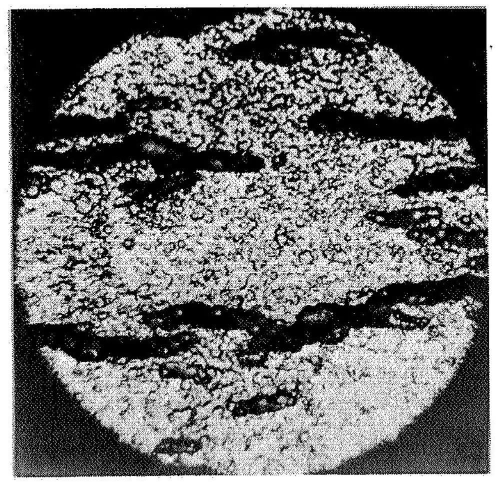
In many respects, test results on the NASA and Lockheed specimens were in agreement although some differences were noted. Several conclusions common to both are presented below. Each is followed by a brief discussion which describes similarities and differences in these two sets of data.

1. For both Lockheed and NASA specimens exposed at 600°F, cracking damage in the 6-4 alloy is not as severe as in the 8-1-1 alloy.

    NASA test data indicates that this result is also true for specimens exposed at 700°F, but the Lockheed 700°F tests did not show a consistent difference in cracking susceptibility for these two alloys. This lack of agreement between NASA and Lockheed 700°F test results may be due to the different nature of the information provided by these two test techniques. As previously discussed, the NASA method provides information on the mechanical strength of a cracked specimen, whereas the Lockheed method provides information on the numbers and/or lengths of the cracks which have formed. This may help explain the fact that the Lockheed test results demonstrated a strong dependence on salt loss, whereas no such correlation is evident in the NASA test results. For example, salt removal produces regions which are essentially devoid of cracks, while other nearby regions may be severely damaged. This reduces the crack count value significantly, but may not greatly affect the residual strength of a sample since those regions weakened by cracks would provide sites susceptible to mechanical failure.

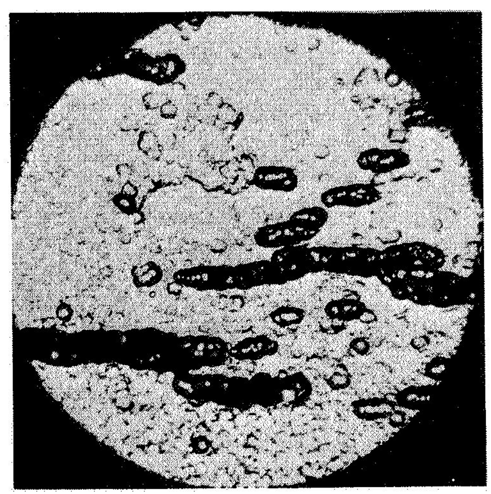


Tunnel exposure 15 hr at 700°F
Ti-6Al-4V, spec. no. 19



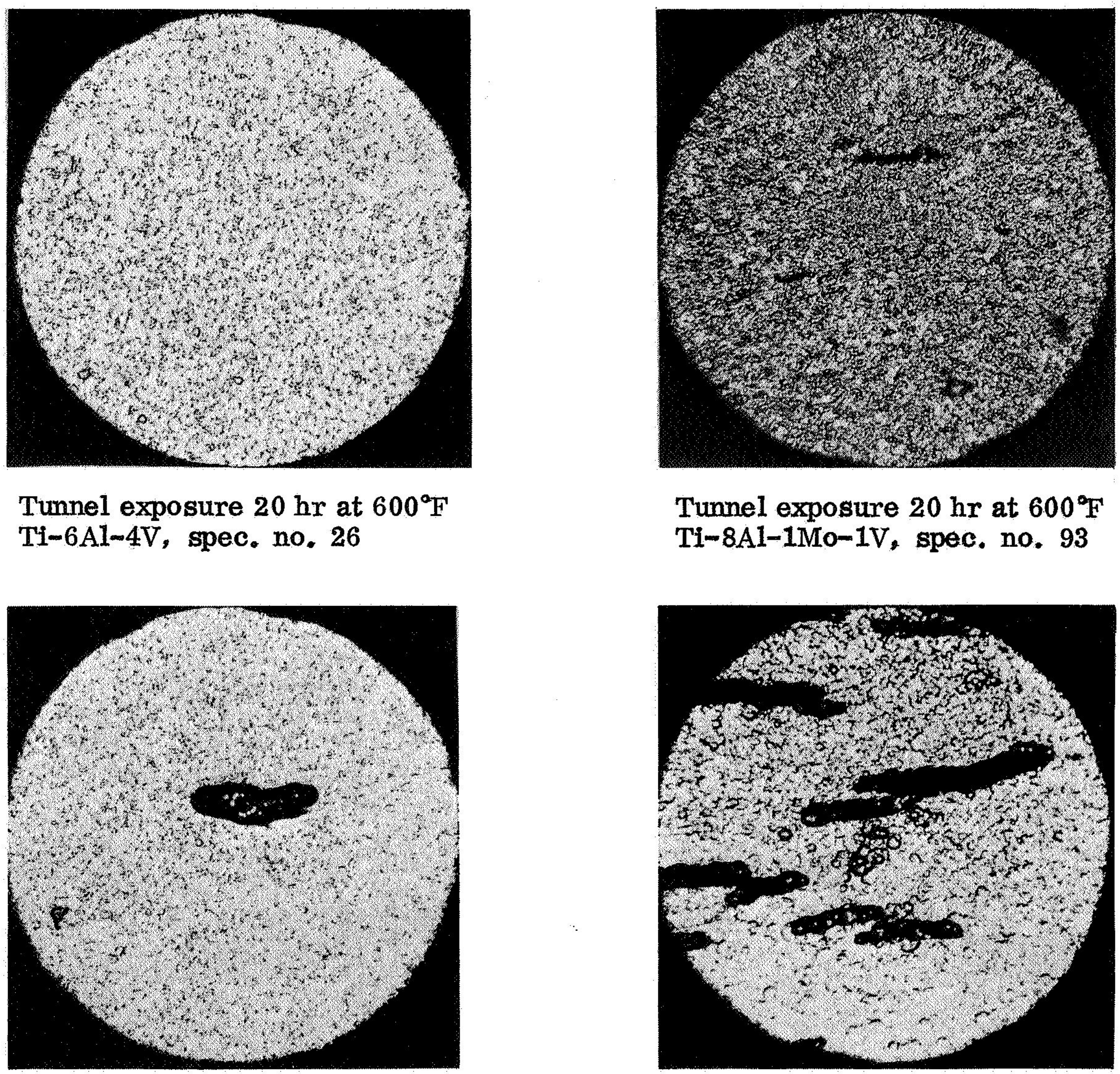
Tunnel exposure 15 hr at 700°F
Ti-8Al-1Mo-1V, spec. no. 11

Av. temp. oven 15 hr at 700°F
Ti-6Al-4V, spec. no. 5



Av. temp. oven 15 hr at 700°F
Ti-8Al-1Mo-1V, spec. no. 6

Figure 5-19 SURFACE PHOTOMICROGRAPHS OF NASA SPECIMENS - 70X



Tunnel exposure 20 hr at 600°F
Ti-6Al-4V, spec. no. 26

Tunnel exposure 20 hr at 600°F
Ti-8Al-1Mo-1V, spec. no. 93

Av. temp. oven 20 hr at 600°F
Ti-6Al-4V, spec. no. 31

Av. temp. oven 20 hr at 600°F
Ti-8Al-1Mo-1V, spec. no. 56

Figure 5-20 SURFACE PHOTOMICROGRAPHS OF NASA SPECIMENS - 70X

Another factor which could contribute to differences in NASA test results in these two alloys may stem from the fact that the basic mechanical properties of the 8-1-1 alloy are somewhat different than for the 6-4 alloy. The 8-1-1 alloy in general is more notch sensitive and the residual strength of this material may therefore be more sensitive to the presence of cracks than the 6-4 alloy. Furthermore, the fact that the 8-1-1 alloy had somewhat higher stress levels than the 6-4 alloy may also have contributed to these differences.

2. In the 600°F tests, cracking damage in both Lockheed and NASA oven test specimens was more extensive than in the tunnel test specimens.

   Lockheed tests indicated that this was also generally true in the case of the 700°F test specimens (particularly for the 8-1-1 alloy), but 700°F NASA specimens did not show this difference. Again this apparent discrepancy may be due to the effects of salt removal and basic differences in the test method discussed above.

3. In the 700°F tests, the majority of the cracking damage on these specimens occurred during the first 15 hours of exposure.

   This observation is evident from the NASA test results since longer exposure times did not produce further cracking damage. In the case of the Lockheed specimens, this result is not as obvious since many of the longer exposure specimens suffered extensive salt removal which produced specimens with lower crack counts. However, since it is obvious that longer times cannot produce fewer cracks, it can only be deduced that the salt on the Lockheed specimens became loosened early in each test and that essentially all of the cracking damage took place during the initial portion of the exposure. The fact that longer time exposure specimens suffered more salt removal may have been largely due to the particular geometric location of the specimen in the mounting plate. Specimens in the center tended to lose more salt (possibly from shock interactions) and those were the same specimens subjected to long exposure times.

4. For both Lockheed and NASA oven specimens the extent of the cracking which occurred in the two different types of oven tests (i.e., the constant average temperature oven tests and the directly corresponding oven tests) was not significantly different.

   This result is not too surprising since there were actually only small differences in the exposure temperatures and times utilized in the two types of oven tests.

5. The extent of the cracking indicated on the photomicrographs of the NASA test specimens is in qualitative agreement with the Lockheed specimen test results.

   Photomicrographs of the NASA specimens (Figs. 5-19 and 5-20) provide information on the relative extent of cracking which is essentially in agreement with the Lockheed test results. For example, qualitative comparison of NASA photomicrographs with the Lockheed cracking data (Table 5-2) reveals that for both types of 700°F specimens, the extent of cracking was similar for both the 8-1-1 and the 6-4 alloy specimens, however, in the 600°F tests cracking was considerably less severe in the 6-4 than in the 8-1-1 alloy. This qualitative agreement suggests that the differences in surface preparation for the two types of specimens (i.e., polished vs. unpolished) do not greatly affect these test results.

In view of the above observations, it can be stated that the rapid air movements and reduced pressures characteristic of supersonic flight environments do not eliminate the titanium hot salt stress corrosion problem. In general however, oven test conditions tend to produce more severe cracking than supersonic environments, especially in the lower temperature regions. This indicates that the large amount of existing oven test data provides conservative hot salt cracking information to the designer of supersonic aircraft.

Since the differences between supersonic and laboratory testing are more pronounced at lower temperatures, future tunnel tests should be run on various alloys in the 450-550°F range. This lower temperature region is more representative of the anticipated temperature conditions of near-future supersonic aircraft and would be of greater value to the designer. In this program, the temperatures were deliberately maintained at high values to ensure that sufficient cracking would occur to allow valid comparison between laboratory oven and supersonic tunnel test data.

APPENDIX A

STRESS LEVEL DETERMINATION FOR LOCKHEED BEND SPECIMENS

Consider a thin strip of sheet metal constrained and buckled between two rigid limits a distance D apart. There is a horizontal restoring force P which produces a maximum deflection δ of the neutral fiber (Fig. A1).

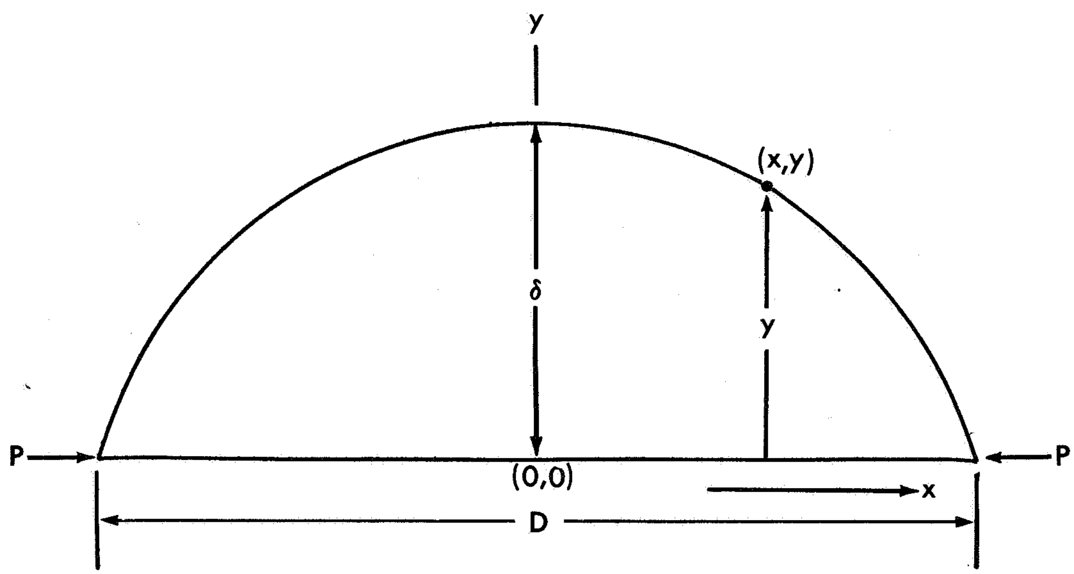


Figure A1 DIAGRAM OF A SPECIMEN IN THE BUCKLED CONFIGURATION

The differential equation which describes this static system was the first derived by Euler by means of a moment balance. This equation has the form:

$$\frac{\dfrac{d^2y}{dx^2}}{\left[1 + \left(\dfrac{dy}{dx}\right)^2\right]^{3/2}} + \frac{P}{EI}y = 0 \qquad (1)$$

where E is the elastic modulus of the material and I is the moment of inertia of the cross sectional area.

Equation (1) may be solved rigorously yielding a number of important mathematical relationships involving the outer fiber stress levels as a function of position along the specimen. The basic assumptions made in deriving these equations are given below: (1) the stress-strain relationship for the material may be described by a linear function (Hooke's Law), (2) the compressive modulus for the material has the same value as the tensile modulus, and (3) anticlastic bending effects are not considered.

One of the most useful equations derivable from equation (1) describes the outer fiber stress level at any y coordinate (Refer to Fig. 2-1); this equation has the form

$$S = \frac{q^2 E t}{3D^2 u^2} (6y - t \cos \psi) \tag{2}$$

where t is the specimen thickness

- u is a dimensionless parameter and is given by the ratio $u = \delta/D$
- q is another dimensionless parameter related to u by means of a function involving elliptic integrals. (The relationship between q and u has been determined and is given in Table A1)
- ψ is the slope angle at the coordinate y and is given by $\psi = \tan^{-1} \left|\frac{dy}{dx}\right|$

In equation (2), tensile stress is designated as a positive quantity. The negative term in the right side of the brackets is a result of the component of the compressive restoring force P which acts (in the tangential direction) on the cross sectional area of the specimen. The negative term is small but does become important in determining the stress levels near the ends of the specimen where the height coordinate y is small. In practice the $\cos \psi$ is often very nearly unity since the slope angle is usually small. The use of $\cos \psi = 1$ over the entire length of the specimen will usually result in negligible error in stress determination.

In principle, equation (2) could be used as it stands for stress determination, but use in this form would require experimental measurement of the coordinate y at each point along the specimen where it is desired to calculate the stress. In actual practice it is much more convenient to use equation (2) in conjunction with several other equations. One such relationship involves the length of the specimen L in terms of the dimensionless parameter u . The rigorous relationship between the quantities L and u involves elliptic integrals, but this has been determined and is presented in Table A-2 in dimensionless form.

Another extremely useful relationship may be derived by introducing some very valid approximations into equation (2). This is accomplished by

TABLE A1

u	q
0.0000	0.0000
0.06414	0.1000
0.09141	0.1414
0.1131	0.1737
0.1369	0.2079
0.1418	0.2149
0.1490	0.2250
0.1560	0.2346
0.1612	0.2419
0.1737	0.2588
0.2396	0.3420
0.3465	0.4546
0.4003	0.5000
0.6431	0.6428
0.8818	0.7660
1.1340	0.8660

TABLE A2

$\frac{L}{D}$	u
1.0000	0.0000
1.0101	0.06414
1.0205	0.09141
1.0312	0.1131
1.0459	0.1369
1.0488	0.1418
1.0537	0.1489
1.0587	0.1559
1.0627	0.1612
1.1348	0.2396
1.3494	0.4003
1.7876	0.6431
2.2281	0.8818
2.8238	1.1340

realizing that the actual shape of the x - y curve is closely approximated by a cosine function

$$y \simeq D u \cos \frac{\pi x}{D} \tag{3}$$

and thus the approximate slope angle ψ may be determined from the absolute value of the first derivative of (3), or

$$\psi \simeq \tan^{-1}\left(\pi u \sin \frac{\pi x}{D}\right) \tag{4}$$

Appropriate substitution yields a very close approximation of equation (2)

$$S \simeq \frac{q^2 E t}{3 D^2 u^2} \left(6 D u \cos \frac{\pi x}{D} - t \cos \psi\right) \tag{5}$$

Equation (5) is a very convenient relationship. To determine outer fiber stress one simply needs to know the length of the specimen and the span distance D. From Table A2 the corresponding value of u is determined which in turn defines a corresponding quantity q from Table A1. Then by using equation (5) the stress at any x coordinate may be determined. Usually the $\cos \psi$ will be sufficiently close to unity to allow this additional simplification, and many times the entire negative compressive stress term may be neglected without significant loss of accuracy.

A minor difficulty may arise from the fact that cracking and corrosion data are usually gathered from flattened specimens and are expressed in terms of the distance along the specimen instead of the x coordinate. This problem may be eliminated by realizing that the distance along the specimen L_A (from the center) is very nearly a linear function of the x coordinate, thus,

$$x \simeq \frac{D}{L} L_A \tag{6}$$

The rigorous relationship between x and L_A involves elliptic integrals but for all practical purposes, equation (6) may be used without significant loss of precision.

Still another problem arises as a result of the fact that some stress relief may occur when a specimen is exposed to an elevated temperature environment. This problem usually only arises in tests at higher temperatures and may generally be circumvented by maintaining the stress levels in the specimens at relatively low values. However, when it is evident that relaxation has occurred, appropriate allowances should be made. Stress relaxation causes the stress levels to be gradually reduced during the test. Any particular point along the specimen will experience a range of stress levels ranging from a higher initial value to a somewhat lower value at the termination of the exposure.

In this program relatively high temperatures and stress levels were utilized, and noticeable stress relief did take place. For simplicity, it was assumed that the stress relief occurred in a particular distribution over the surface of the specimen in accordance with the cosine type relationship of equation (5). The actual stress relaxation levels which occurred were deduced from the maximum permanent deflection which occurred in the unloaded specimen after testing. This deflection value was converted to the corresponding dimensionless parameter u and used in conjunction with equation (5) to define the stress relief distribution. Then at each point of interest along the specimen this stress relief term was subtracted from the calculated initial stress level, thus providing a means of estimating the true stress level at the termination of the test. It could be argued that an average stress level, intermediate between initial and final values should be used to describe the stress at any particular point. However, the final stress at any point will be the lowest and therefore most conservative estimate of stress level. The conservative approach was adopted for this study and all stress level data presented in this report have been corrected in this manner to allow for stress relief effects.

The method used to prepare and analyze specimens is summarized below in the following steps:

1. Decide on a nominal specimen size, and define a holder dimension D. Select several arbitrary values of length L slightly in excess of D and determine the corresponding maximum stress levels utilizing Tables A1 and A2 in conjunction with equation (5) evaluated at its maxima ($x = 0$). Prepare a plot of maximum stress as a function of the L/D ratio. Select a desired maximum stress level and cut a specimen to the required length as determined from this plot.

2. Insert the specimen in the holder slot and experimentally determine the distance between the centers of the ends of the mounted specimen. Due to such factors as thickness and holder geometry, the centers of the ends of the specimen may not exactly correspond to the slot dimension D. Therefore, it is necessary to empirically determine this value and designate it is the true value of D. If all holders have similar geometry and all specimens have similar thicknesses, the true value of D will not vary significantly from one specimen to the next even though various lengths and stress levels may be employed. In most cases a single corrected value of D may be used for all test specimens.

3. Using the corrected value of D , calculate a new length from the original required L/D ratio determined in step 1. Prepare specimens to this corrected length and mount in holders.

4. After testing, associate cracking data obtained from flattened sheet to the x coordinate by means of equation (6).

5. Relate cracking data (by means of x coordinate) to outer fiber stress levels using equation (5).

REFERENCES

1. Heimerl, G. J.; Braski, D. N.; Royster, D. M.; and Dexter, H. B.: Salt Stress Corrosion of Ti-8Al-1Mo-1V Alloy Sheet at Elevated Temperatures. Stress Corrosion Cracking of Titanium, ASTM STP 397, Am. Soc. Testing Mats., 1966, pp. 194-212.

2. Heimerl, G. J.; and Braski, D. N.: A Stress Corrosion Test for Structural Sheet Materials. Materials Research and Standards, vol. 5, no. 1, Jan. 1965, pp. 18-22.

3. Baker, H. D.; Ryder, E. A.; and Baker, N. H.: Temperature Measurement in Engineering. Volume I. John Wiley & Sons, Inc., 1953.

4. Braski, D. N.; and Heimerl, G. J.: The Relative Susceptibility of Four Commercial Titanium Alloys to Salt Stress Corrosion at 550°F. NASA TN D-2011, 1963.

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